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# V/STOL TILT ROTOR AIRCRAFT STUDY

## VOLUME VI

## PRELIMINARY DESIGN
## OF A COMPOSITE WING FOR TILT
## ROTOR RESEARCH AIRCRAFT

MARCH 1973

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BOEING CENTER
P. O. Box 16858
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for

Ames Research Center
National Aeronautics and Space Administration
and
United States Army Air Mobility Research and Development Laboratory
Ames Directorate



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## FOREWORD

This report is one of a series prepared by The Boeing Vertol Company, Philadelphia, Pennsylvania for the National Aeronautics and Space Administration, Ames Research Center, Moffett Field, California under contract NAS2-6598. The studies reported under Volumes I through IV and VIII through X were jointly funded by NASA and the U.S. Army Air Mobility Research and Development Laboratory, Ames Directorate. Volumes V through VII were funded by the U.S. Air Force Flight Dynamics Laboratory, Wright Patterson Air Force Base, Ohio.

This contract was administered by the National Aeronautics and Space Administration. Mr. Richard J. Abbott was the Contract Administrator, Mr. Gary B. Churchill, Tilt Rotor Research Aircraft Project Office, was the Technical Monitor, and coordination and liaison with the U.S. Air Force Flight Dynamics Laboratory was through Mr. D. Fraga.

The complete list of reports published under this contract is as follows:

Volume I -- Conceptual Design of Useful Military and/or Commercial Aircraft, NASA CR-114437

Volume II -- Preliminary Design of Research Aircraft, NASA CR-114438

Volume III -- Overall Research Aircraft Project Plan, Schedules, and Estimated Cost, NASA CR-114439

Volume IV -- Wind Tunnel Investigation Plan for a Full Scale Tilt Rotor Research Aircraft, CR-114440

Volume V -- Definition of Stowed Rotor Research Aircraft, NASA CR-114598

Volume VI -- Preliminary Design of a Composite Wing for Tilt Rotor Aircraft, NASA CR-114599

Volume VII -- Tilt Rotor Flight Control Program Feedback Studies, NASA CR-114600

Volume VIII -- Mathematical Model for a Real Time Simulation of a Tilt Rotor Aircraft (Boeing Vertol Model 222), NASA CR-114601

Volume IX -- Piloted Simulator Evaluation of the Boeing Vertol Model 222 Tilt Rotor Aircraft, NASA CR-114602

Volume X -- Performance and Stability Test of a 1/4.622 Froude Scaled Boeing Vertol Model 222 Tilt Rotor Aircraft (Phase 1), NASA CR-114603

# ABSTRACT

This report presents the results of a study of the use of composite materials in the wing of a tilt rotor aircraft. An all-metal Search and Rescue (SAR) tilt rotor aircraft was first defined to porvide a basis for comparing composite with metal structure. A configuration study was then done in which the wing of the metal aircraft was replaced with composite wings of varying chord and thickness ratio. The results of this study defined the design and performance benefits obtainable with composite materials. Based on these results the aircraft was resized with a composite wing to extend the weight savings to other parts of the aircraft. A wing design was then selected for detailed structural analysis. A development plan including costs and schedules to develop this wing and incorporate it into a proposed flight research tilt rotor vehicle has been devised.

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## 1.0 SUMMARY

This report presents the results of a study conducted by the Boeing Vertol Company to define the effects of the use of composite materials in the wing of a tilt rotor aircraft. The objectives of the study were:

1. to define the design and performance benefits obtained with composite materials

2. to design a composite wing for the tilt rotor research aircraft

3. to establish a development plan for a composite wing for the tilt rotor research aircraft

The USAF SAR aircraft described in Reference 1 was used as a basis for the study. Since this aircraft was designed with some composite structure, it was resized to an all-metal configuration. The all-metal tilt rotor aircraft was used as a basis for comparison.

To determine the optimum wing configuration, two parametric trade studies were conducted. In the first wing chord was held constant and thickness was varied. In the second thickness ratio was held and chord was varied. These studies were detailed enough to show the effects of changing chord and thickness on the drag and weight of the wing and particularly to show the cross over point between the strength critical and stiffness critical design conditions.

Three design point aircraft were defined for purposes of comparison. The first is the all-metal reference aircraft described above. The second is a resized composite wing aircraft in which composite materials were used for the entire wing and the resizing was done to extend the weight savings to other parts of the aircraft. The third design point is simply the all-metal aircraft with a composite wing. In this case the wing weight saving was taken as a payload or performance benefit.

Gross weights and rotor diameters of these three aircraft are compared with the aircraft of Reference 1 in the following table:

| Aircraft | Rotor Dia. Feet | Design GW Lbs | Δ Weight Lbs | Δ Weight % |
|---|---|---|---|---|
| Reference 1 (Moderate use of composites throughout) | 27.0 | 16,970 | -1,055 | -6.2 |
| All - metal | 28.9 | 18,025 | ----- | --- |
| All - metal plus max use of composites in wing only | 28.9 | 17,650 | -375 | -2.1 |
| All - metal plus composite wing - resized | 27.1 | 17,242 | -783 | -4.3 |

It may be noted that on the third aircraft (composite wing only, not resized) the only weight saving is the 30% reduction in wing weight from 1,250 to 875 pounds.

A wing configuration was chosen for further study and a design and stress analysis done. A simple two-spar configuration was chosen for the wing torque box with a spanwise well in the upper surface for the cross shaft. The torque box is a honeycomb shell consisting of Boron-Epoxy facings on a fiberglass honeycomb core.

A development plan has been devised which considers the design, construction, and testing of a composite wing for the tilt rotor research aircraft. In order to arrive at a minimum cost program, only the main spar torque box is built in composites for this program. The auxiliary surfaces (flaps, umbrellas, etc.) are existing metal components.

## 2.0 INTRODUCTION

In March 1972, the Boeing Vertol Company completed a study of tilt rotor aircraft under the joint sponsorship of NASA and the U.S. Army (References 1 - 4). Part of that study (Reference 1) covered the conceptual design of useful military and civil tilt rotor aircraft for the 1975-1980 time period. Composite materials were utilized as a means of reducing airframe structural weight. In that study the weight factor for composites was taken as 15%. Design studies and prototype test data have indicated, however, that larger savings could be realized with present technology.

Consequently, the Boeing Company was asked by the U.S. Air Force Flight Dynamics Laboratory, through an add-on to the NASA contract, to investigate the use of composite materials in the wing of a tilt rotor aircraft. The objectives of this study were as follows:

1. to define the design and performance improvements a composite wing provides for tilt rotor aircraft

2. design a composite wing for the tilt rotor SAR aircraft

3. establish a development plan for a composite wing for the tilt rotor research aircraft

This report presents the results of the study. The preliminary design studies required to define the optimum wing configuration are described in Section 3. The advanced design of a composite wing is described in Section 4 and the development plan in Section 5.

# 3.0 PRELIMINARY DESIGN STUDIES

## 3.1 INTRODUCTION

A study was performed to show the potential benefits obtainable from the application of composite materials to an advanced type of VTOL aircraft - the tilt rotor. A promising operational application for the tilt rotor configuration - a USAF search and rescue (SAR) mission - was chosen for this study. This application for the tilt rotor concept had previously been studied by Boeing during 1971-72 in a NASA/Army sponsored "V/STOL Tilt Rotor Aircraft Study" (Reference 1). Moderate application of composites to the fuselage, wing, and empennage was assumed for the aircraft defined in that study.

The present study examines in more detail the application of composites to the wing only with the following objectives:

(1) To show the improvements in mission performance achievable by applying composite materials to the wing alone of an all-metal search and rescue tilt rotor aircraft.

(2) To show the overall weight and size benefits obtainable by resizing the total aircraft structure to take advantage of the reduced wing weight - even though composites were still applied only to the wing.

To provide a basis for comparison, the SAR aircraft of Reference 1 was resized to an all-metal structure.

(For reference, some of the characteristics of the Reference 1 aircraft are tabulated below.)

M222-1F SAR TILT ROTOR (REF. 1)

| | |
|---|---|
| Gross Weight (lb) | 16970 |
| Weight Empty (lb) | 11500 |
| Wing Area (sq. ft.) | 186 |
| Wing Span (ft.) | 34.4 |
| Thickness ratio (t/c) | 21% |
| Rotor Diameter (ft.) | 27.0 |
| Solidity Ratio | .133 |
| Power Plant | (2) Lycoming PLT-27 |
| Rated Power (Shp) | 1950 |

In the remainder of this section, the criteria for aircraft sizing are first discussed (Section 3.2), followed by a description of the all-metal aircraft and its performance (Section 3.3). The effects of applying composite materials to the wing of the all-metal aircraft are then shown in Section 3.4. This study included the variation of wing geometry (thickness and chord) to determine whether secondary benefits could be credited to the use of composites by making changes in the wing geometric design. Section 3.5 discusses the effect of resizing the remainder of the aircraft structure to take advantage of the lighter wing. These preliminary design studies are then summarized in Section 3.6.

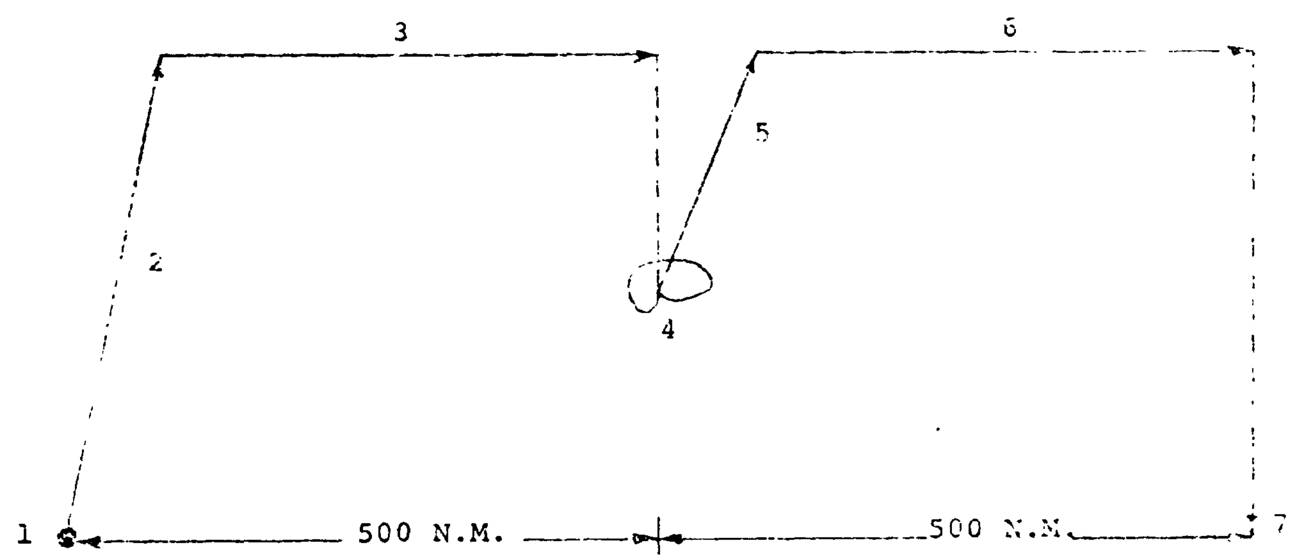
## 3.2 DESIGN CONSIDERATIONS

### 3.2.1 Design Mission Profile

All aircraft were sized to perform a 500 NM Search and Rescue (SAR) mission (Figure 3-1). This is a "HI-HI" mission consisting of a takeoff at SL/95°F, climb to optimum altitude, cruise out at NRP to the 500 NM radius, hover for 1/2 hour at 5000ft/95°F and recover three (3) rescuees, and return without inflight refueling. The optimum cruise altitude (based on minimum fuel) was found to be 20,000 ft.

The aircraft were assumed to carry a four-man crew consisting of two pilots, a crew chief, and a paramedic. The mission load was specified at 150 lb of rescue equipment (litters, forest penetrator, rescue sling, et.), airborne electronics and equipment required to locate the rescuee, and a 5.56mm machine gun and ammunition.

The engines, rotors, and drive system were sized by an alternate mission requirement. This was that the aircraft be capable of hovering at the mission midpoint at T/W=1.1 with a total of seven rescuees - the additional four rescuees being the crew of a downed sister ship. It was assumed that inflight refueling would be allowed under these conditions so that the mission fuel requirement is determined by the basic mission shown in Figure 3-1.




1. WARM UP, TAXI AND TAKEOFF: 3 MIN. @ NORMAL RATED POWER, SEA LEVEL, 95°F
2. CLIMB TO OPTIMUM ALTITUDE @ MILITARY POWER AND SPEED FOR MAXIMUM RATE OF CLIMB
3. CRUISE OUTBOUND @ NORMAL RATED POWER SPEED
4. HOVER 1/2 HR., EFFECT RESCUE OF 3 PEOPLE (600 LBS) @ 5000'/95°F, MILITARY POWER
5. CLIMB TO OPTIMUM ALTITUDE @ MILITARY POWER AND SPEED FOR MAXIMUM RATE OF CLIMB
6. CRUISE INBOUND @ NORMAL RATED POWER SPEED
7. LAND WITH 10% (INITIAL) FUEL RESERVE

NOTES:

1. MISSION FLOWN @ STANDARD ATMOSPHERE CONDITIONS UNLESS OTHERWISE NOTED.
2. SFC INCREASED 5% PER MIL-C-5011A

FIGURE 3-1. DESIGN MISSION PROFILE - SAR HI-HI MISSION

### 3.2.2 Propulsion System

#### 3.2.2.1 Engine Cycle

In the Reference 1 study the Lycoming PLT-27 engine rated at 1950 horsepower was chosen to power the SAR tilt rotor aircraft. This engine met the midpoint hover requirement with a reasonable rotor diameter. This engine has been retained in the present study.

### 3.2.2 Transmission and Rotor Design

The transmissions and rotors were structurally designed by the maximum rated horsepower of the engine at the hover rpm. That is, no transmission torque limits were applied at hover rpm but power was limited to 70% of sea level maximum at cruise rpm.

The rotors considered in the study were assumed to be of the same hingeless design as the rotor defined for the Tilt Rotor Research Aircraft in NASA CR-114438, "Preliminary Design of Research Aircraft", Reference 2. The blades were assumed to be rectangular in planform and to utilize the BV23010-1.58 airfoil outboard of the blade cuff. The same basic design and type of construction was assumed and the same weight factors were used.

### 3.2.3 Aircraft Drag

A simplified drag model was used for sizing the design point aircraft. The model represents the drag of the aircraft as linear functions of wing area. The methods of Boeing Document D8-2194-1, "Drag Estimation of V/STOL Aircraft", Reference 7, were used to calculate the intercept and slope of the trend curve. The drag trend used is shown in Figure 3-2. This curve is identical to that shown for the SAR aircraft in Reference 1.

In the composite wing trade study parametric variations of wing drag with wing chord and thickness ratio were computed. These values were then used to increment the basic $f_e$ of the baseline aircraft. The procedure and drag values used are discussed in more detail in Section 3.4.1.

### 3.2.4 Criteria for Selecting Design Point Aircraft

The design point aircraft were sized to the mission requirements discussed in Section 3.2.1. Additionally, the following design constraints were imposed:

1. Thrust-weight ratio capability at the mission mid-point of at least 1.1 with seven (7) rescuees
2. Maximum hover disk loading of 15 psf
3. Rotor solidity greater than .058
4. Wing chord to rotor diameter ratio of 0.2

In general, these constraints are the result of practical considerations in the design of tilt rotor aircraft. The disk loading limit, for example, was imposed to avoid excessive downwash velocities in hover. Downwash velocity is directly related to disk loading. At high disk loadings, the resulting high downwash velocities would tend to hamper rescue operations.

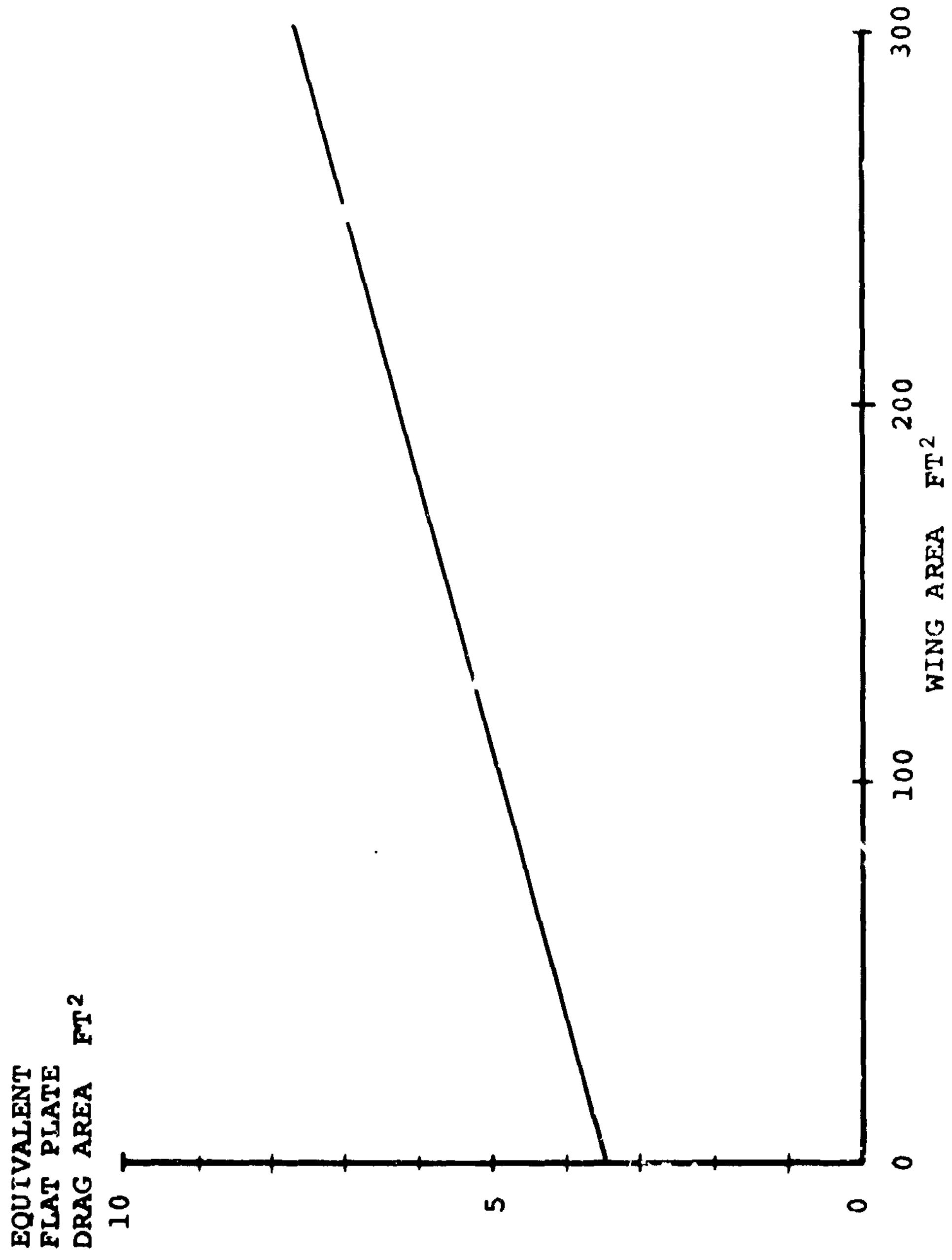



FIGURE 3-2 DRAG TREND

A maximum thrust coefficient-to-solidity ratio, $C_T/\sigma$=0.135, was used, based on stall flutter considerations. However, in no event was the solidity permitted to go below a value of 0.058. The rotor solidity limit is based on practical design and manufacturing considerations related to blade torsional and flapping stiffness requirements. As rotor blades become narrower and thinner at the lower solidities it becomes more and more difficult to tune them and still meet design fatigue life requirements.

The chord-diameter ratio value used is a nominal value selected on the basis of previous design experience. It has been found that C/D=0.2 gives wing aspect ratios that provide adequate control of the static divergence and whirl flutter modes without excessive weight and performance penalties. The effect of chord-diameter ratio on mission performance has been investigated in this study and is discussed in Section 3.4.2.

Fixing chord-diameter ratio fixes wing configuration because span has also been specified as a function of diameter. Thus wing loading is a function of disk loading and rotor diameter becomes the design parameter. Th procedure for sizing the design point aircraft then become a matter of sizing aircraft for a series of rotor diameters and determining the minimum weight configuration corresponding to the most critical of the first three design constraints.

### 3.2.5 Wing Structural Design Criteria

Wing structural design criteria for the study are based on those established for the Model 222 tilt rotor research aircraft in Reference 5. In general the same loading criteria were applied except that the limit load factor was reduced to 2.67 to match that used in the Reference 1 design study. Stress allowables used are based on current Boeing practice for composite materials.

### 3.2.6 Composite Weight Factors

Analytical studies, complemented by actual hardware development, have established a 30 percent weight reduction potential for advanced composite material. A survey paper, "Weight Prediction Techniques and Trends for Composite Material Structure", presented at the 30th annual SAWE meeting in 1971 (Reference 6) identified 21 aerospace structural components made from advanced composites. Further research was done to identify the actual weight savings achieved compared to that predicted by the various analytical studies. The following table is reproduced from Reference 6.

Based on this analysis a weight reduction factor of 30% for a composite wing was used in this study. This reduction was agreed upon by the Air Force early in the study program.

| Component | Material | Weight Saving | Remarks | Contractor |
|---|---|---|---|---|
| F-4 Rudder | Boron/Epoxy | 35% | Including Balance Weights | McDonnell Aircraft Company |
| MCAIR IRAD Stabilator | Boron/Epoxy | 27% | 33% Savings on Refined Design | McDonnell Aircraft Company |
| F-111 Stabilator | Boron/Epoxy | 25% | 3-Ply Minimum Gage | General Dynamics Convair Aerospace-Fort Worth Operation |
| F-111 Wing Tip | Boron/Epoxy | 35%/17% | Unpressurized/ Pressurized | Grumman Aircraft Engineering Corporation |
| F-5 Landing Gear Door | Boron/Epoxy | 29% | | Northrop Corporation, Norair Div. |
| F-100D Wing Skins | Boron/Epoxy | N/A | | North American Rockwell-LAD |
| F-14 Stabilator | Boron/Epoxy | 200 lb | | Grumman Aircraft Engineering Corp |
| A-4 Flap | Boron/Epoxy | 32% | | Douglas Aircraft Company |
| A-4 Flap | Graphite/Epoxy | 47% | | Douglas Aircraft Company |
| A-4 Stabilator | Graphite/Epoxy | 32% | | Douglas Aircraft Company |
| VC-10 Aileron Strut | Graphite/Epoxy | 43% | Including End Fittings | Royal Aircraft Establishment |
| C-5A Leading Edge Slat | Boron/Epoxy | N/A | | Lockheed Georgia Company |
| T-39A Wing Box Section | Boron/Epoxy | 37% | | North American Rockwell-LAD |
| Advanced Composite Wing Structure | Boxon/Epoxy | N/A | | General Dynamics Convair Aerospace-Fort Worth Operation |
| F-5 Leading Edge Section | Graphite/Epoxy | 21% | | Northrop Corporation, Norair Div. |
| F-111 Fuselage | Boron/Epoxy | N/A | | General Dynamics Convair Aerospace-Fort Worth Operation |
| | Graphite/Epoxy | | | |
| | Boron/Aluminum | | | |
| CH-47 Rotor Blade | Boron/Epoxy | N/A | | Boeing Vertol |
| S-61 Tail Rotor | BoronGlass/ Epoxy | | | Sikorsky Aircraft |
| ICBM Reentry Vehicle | Boron/Epoxy | N/A | | General Electric |
| Tubular Struts | Boron/Epoxy | 30% | Including End Fittings | North American Rockwell Space Div |
| Missile Payload Adapter | Boron/Aluminum | 30% | | General Dynamics Convair Aerospace-San Diego Operation |

TABLE 3-1 COMPOSITE HARDWARE DEVELOPMENT

## 3.3 ALL METAL AIRCRAFT

### 3.3.1 Description

To provide a baseline against which to measure the benefits obtainable with composites, the SAR aircraft of Reference 1 was resized to an all-metal configuration. This was necessary because that aircraft had some composite materials in it.

The parametric sizing results are shown in Figure 3-3. The data shown are: disk loading, midpoint thrust-weight ratio, and gross weight. As noted, the aircraft is sized by the midpoint hover requirement. This gave a design gross weight of 18025 lb at a disk loading of 13.8 psf. The characteristics of the aircraft are summarized in the following table.

Design Point All-Metal Aircraft Characteristics

| | |
|---|---|
| Gross Weight (lb) | 18,025 |
| Weight Empty (lb) | 12,380 |
| Aspect Ratio | 6.29 |
| Wing Area ($Ft^2$) | 210.1 |
| Wing Span (Ft) | 36.35 |
| Wing Chord (Ft) | 5.78 |
| Taper Ratio ($\lambda$)/Sweepback ($\Lambda$) | 1.0/0 |
| Wing Thickness (%) | 21 |
| Wing Loading ($Lb/Ft^2$) | 85.8 |
| Rotor Diameter (Ft) | 28.9 |
| Chord to Diameter Ratio | 0.200 |
| Rotor Solidity Ratio | .08 |
| Disk Loading ($Lb/Ft^2$) | 13.8 |
| Design $C_T/\sigma$ | .135 |
| Power Plant | (2)Lycoming PLT-27 |
| Rated Power @ SL/STD (SHP) | 1950 |

A summary weight statement for the aircraft is presented in Table 3-2.

### 3.3.2 Performance

The performance characteristics of the all-metal aircraft are

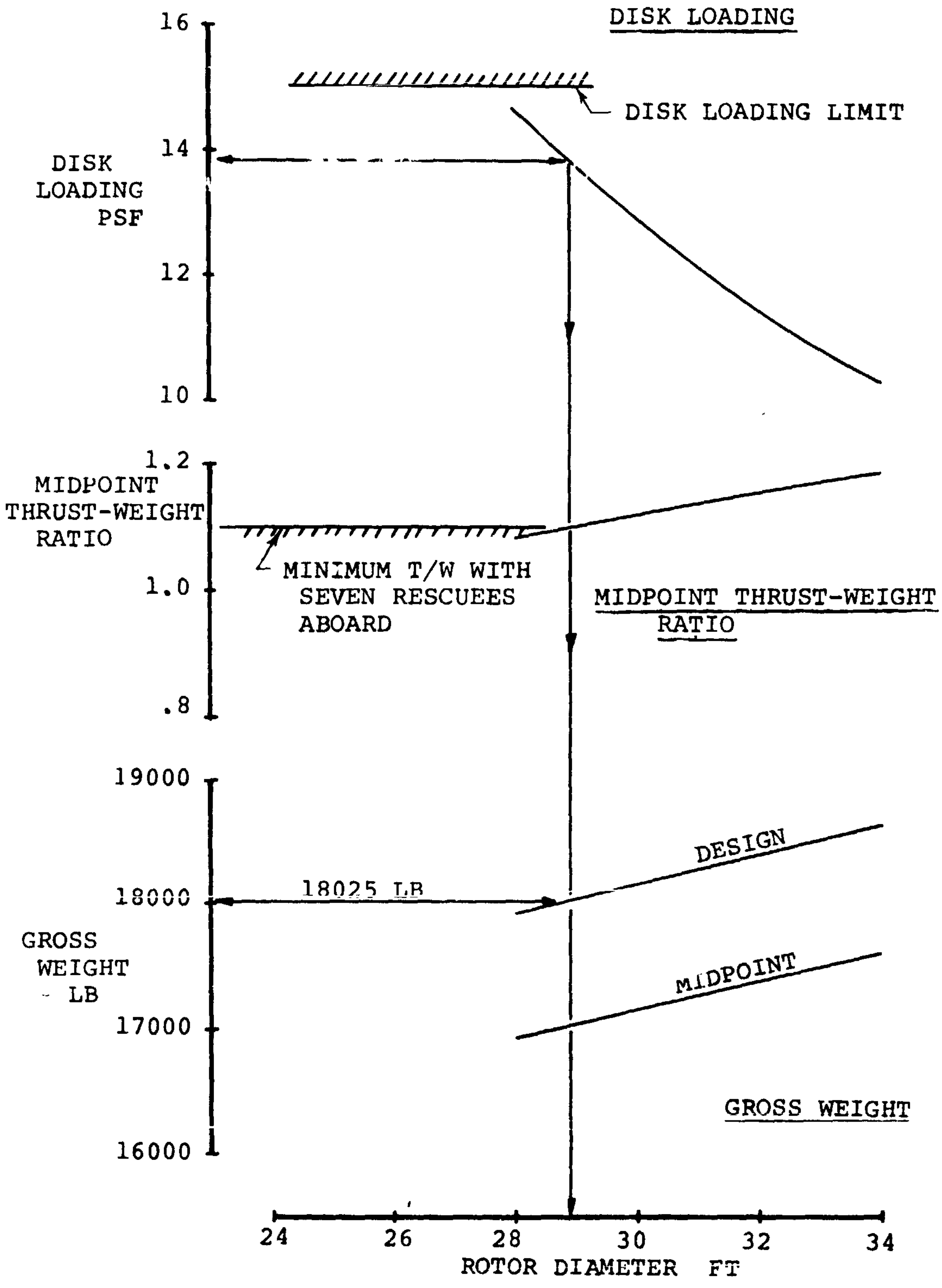



FIGURE 3-3. ALL-METAL SAR TILT ROTOR PARAMETRIC SIZING RESULTS

## TABLE 3-2: SUMMARY WEIGHT STATEMENT USAF - SAR CURRENT MAT'L. & TECH.

| | | | | | | |
|---|---|---|---|---|---|---|
| ENG H.P. EA<br>ROTOR DIA/σ<br>WING AREA | 1950<br>28.9'/.08<br>210 $FT^2$ | | | | | |
| ROTOR GROUP | 1203 | | | | | |
| WING GROUP | 1250 | | | | | |
| TAIL GROUP | | | | | | |
| BODY GROUP | | | | | | |
| BASIC | | | | | | |
| SECONDARY | 2300 | | | | | |
| SECOND.-DOORS, ETC. | | | | | | |
| ALIGHTING GEAR | | | | | | |
| FLIGHT CONTROLS | 1385 | | | | | |
| ENGINE SECTION | 350 | | | | | |
| PROPULSION GROUP | (2692) | | | | | |
| ENGINES(S) | 620 | | | | | |
| AIR INDUCTION | | | | | | |
| EXHAUST SYSTEM | 200 | | | | | |
| COOLING SYSTEM | | | | | | |
| LUBRICATING SYSTEM | | | | | | |
| FUEL SYSTEM | 445 | | | | | |
| ENGINE CONTROLS | | | | | | |
| STARTING SYSTEM | | | | | | |
| PROPELLER INST. | | | | | | |
| *DRIVE SYSTEM | 1427 | | | | | |
| | | | | | | |
| AUX. POWER PLANT | | | | | | |
| INSTR. AND NAV. | 135 | | | | | |
| HYDR. AND PNEU. | 130 | | | | | |
| ELECTRICAL GROUP | 800 | | | | | |
| ELECTRONICS GROUP | 1400 | | | | | |
| ARMAMENT GROUP | 175 | | | | | |
| FURN. & EQUIP. GROUP | 350 | | | | | |
| PERSON. ACCOM. | | | | | | |
| MISC. EQUIPMENT | | | | | | |
| FURNISHINGS | | | | | | |
| EMERG. EQUIPMENT | | | | | | |
| AIR COND. & DE-ICING | 100 | | | | | |
| PHOTOGRAPHIC | | | | | | |
| AUXILIARY GEAR | 110 | | | | | |
| | | | | | | |
| | | | | | | |
| MFG. VARIATION | | | | | | |
| WEIGHT EMPTY | 12380 | | | | | |
| FIXED USEFUL LOAD | | | | | | |
| CREW (4) | 860 | | | | | |
| TRAPPED LIQUIDS | 40 | | | | | |
| ENGINE OIL | | | | | | |
| MISSION EQUIP. | 150 | | | | | |
| FUEL | 4450 | | | | | |
| CARGO | | | | | | |
| PASSENGERS/TROOPS | | | | | | |
| GUN & AMMO | 145 | | | | | |
| GROSS WEIGHT | 18025 | | | | | |

* INCLUDES ——— LB. XMNS. OIL

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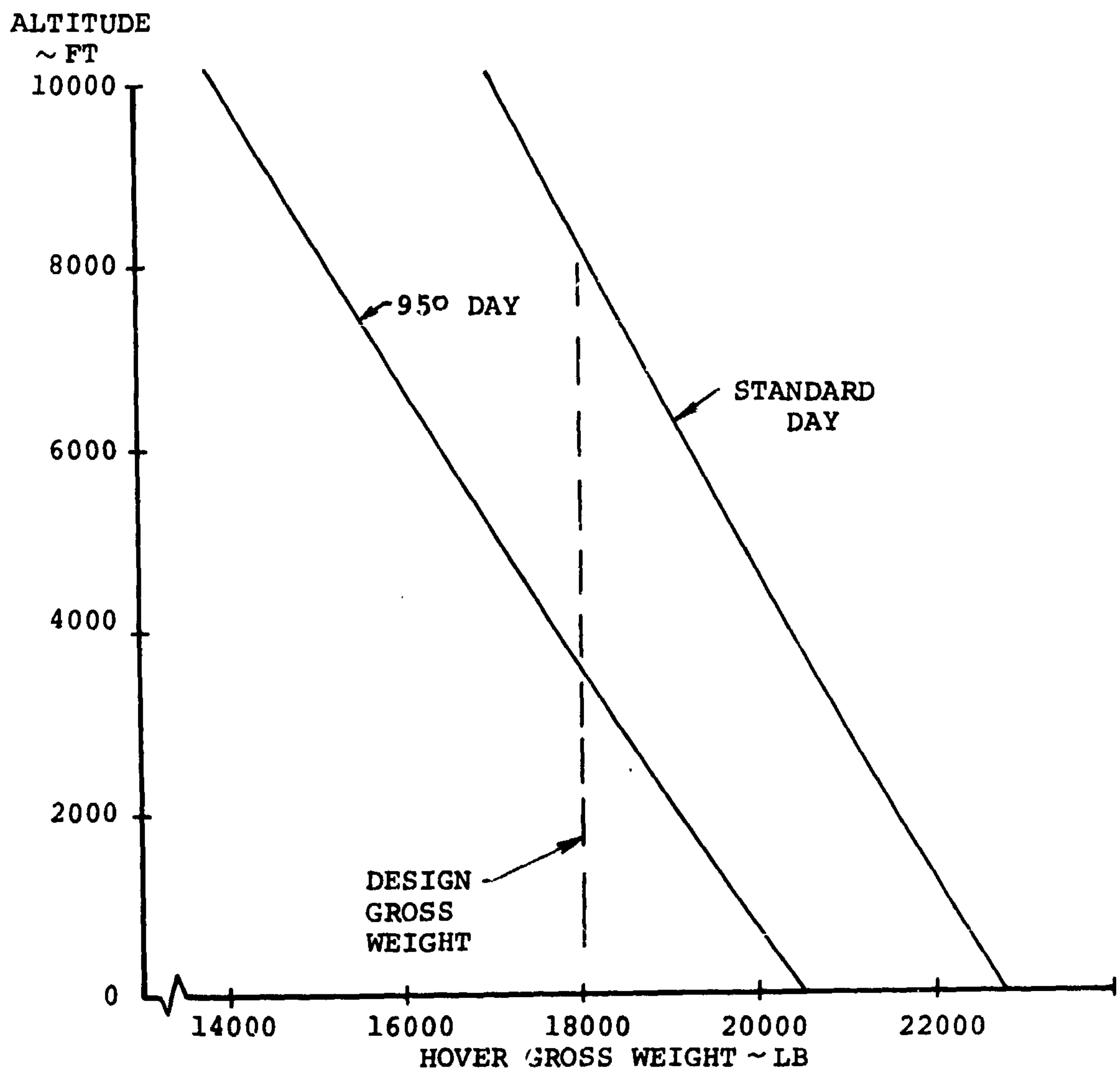
summarized in Figures 3-4 and 3-5. The data are presented for 18025 lb gross weight and include hover ceiling, flight envelope, and climb characteristics.

The aircraft can hover at its design gross weight at about 3600 ft on a hot day (95°F) and over 8000 ft under standard day conditions. (Figure 3-4) These data are based on a thrust-weight ratio of 1.1 which allows 5% margin for download and 5% for maneuverability.

Cruise mode performance is summarized in Figure 3-5. The aircraft is capable of 320 kt at normal power up to 5000 ft and can exceed 300 KTAS up to 17000 ft. The aircraft has adequate climb performance and has absolute ceilings in excess of 25000 ft.




NOTES:

1. T/W = 1.1
2. Military Power
3. Rotor Tip Speed: 750 FPS
4. Design Gross Weight: 18025 LB

FIGURE 3-4. ALL-METAL SAR TILT ROTOR OGE HOVER CEILING

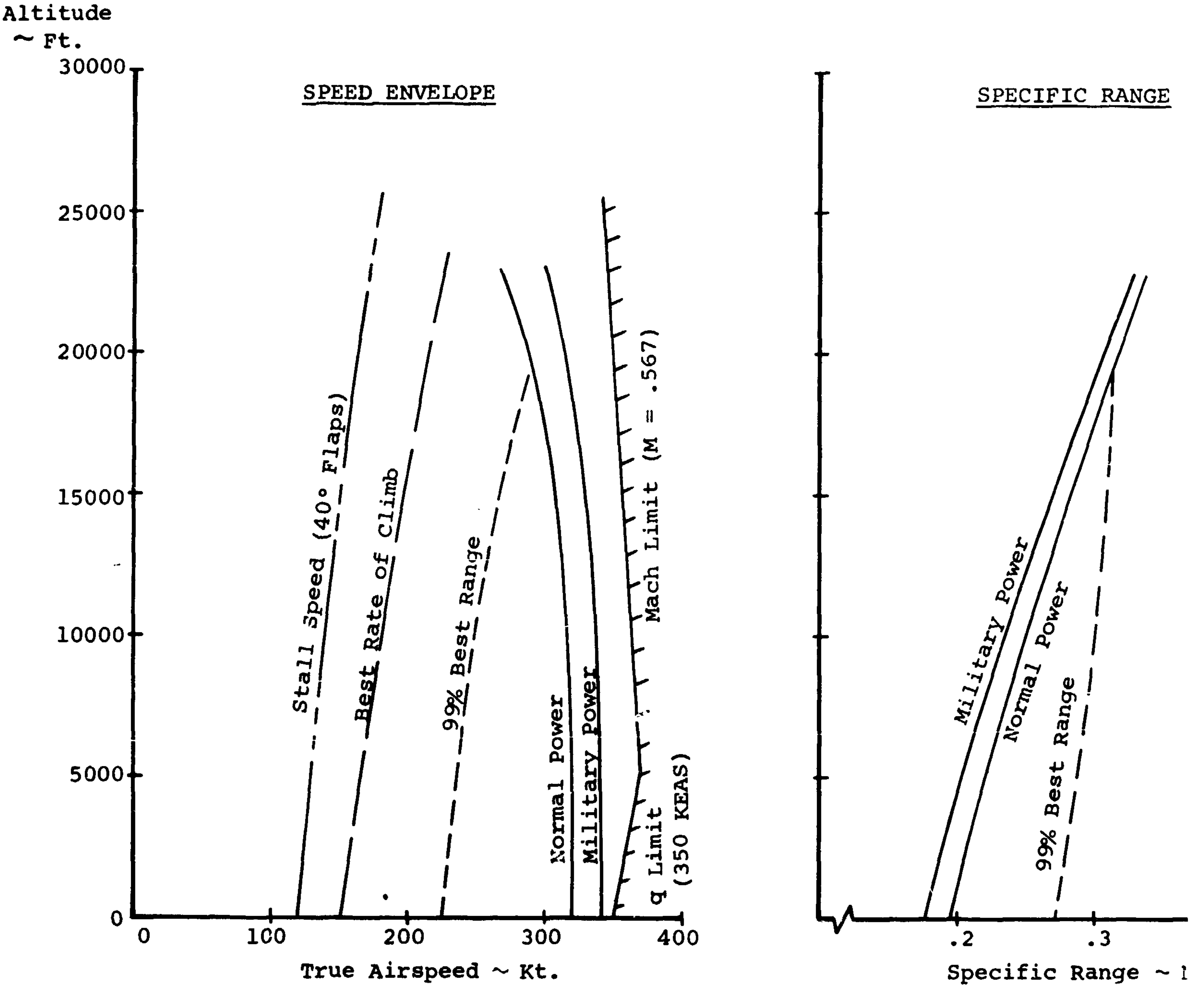
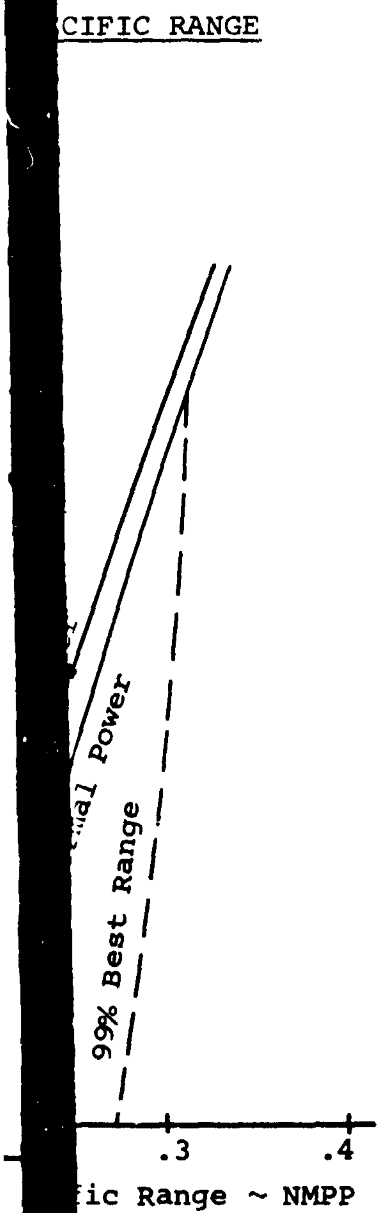
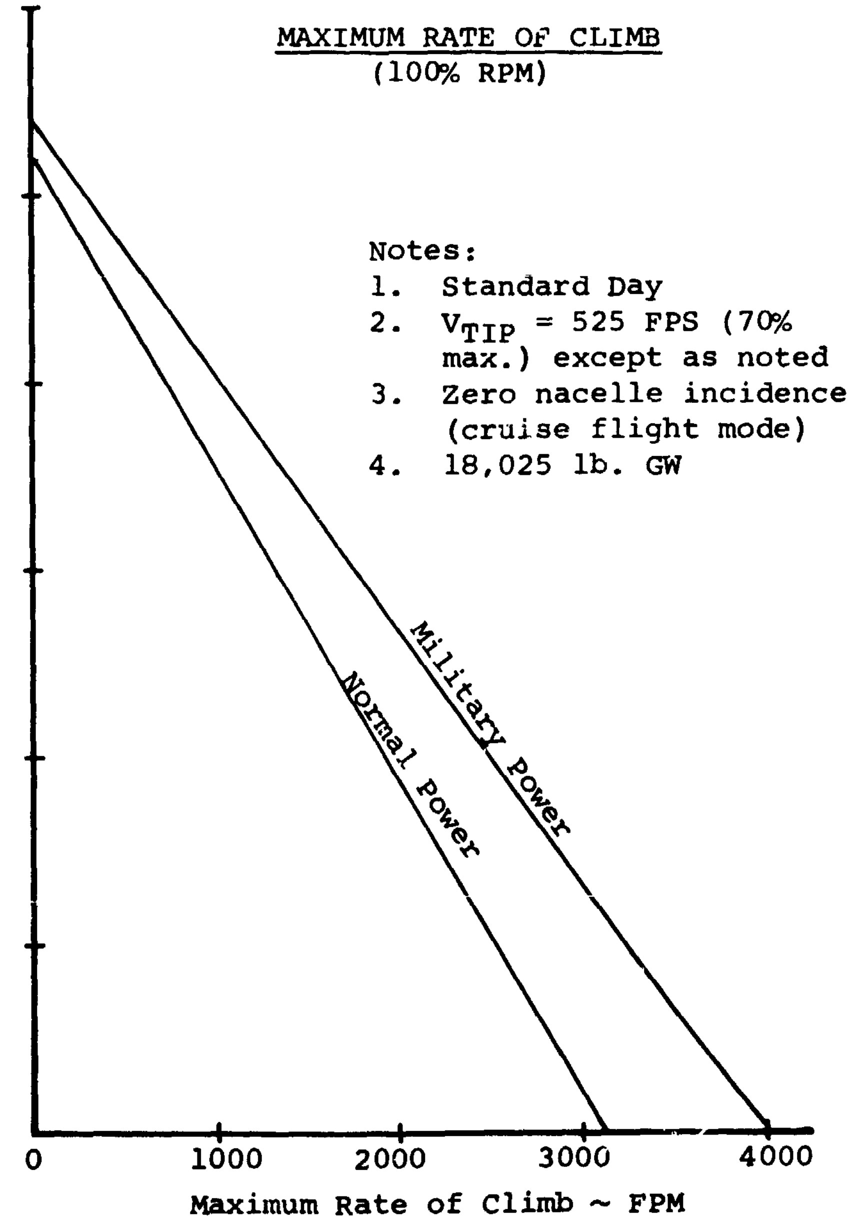



Figure 3-5. All-Metal SAR Tilt Rotor Cruise P
Summary



FOLDOUT FRAME




tor Cruise Performance

FOLDOUT FRAME 2

## 3.4 COMPOSITE WING CONFIGURATION STUDIES

The first objective of the study was to determine the performance benefits obtainable with composites as affected by wing geometry. This was done by replacing the metal wing of the all-metal aircraft (Section 3.3) with composite wings and varying geometry and then computing the mission performance of the modified aircraft.

Wing geometry was varied in two ways: in the first the baseline wing planform was used and thickness ratio was varied from 15% to 24% - the characteristics of these wings are tabulated as follows:

| | |
|---|---|
| chord | - 5.78 ft |
| span | - 36.33 ft |
| aspect ratio | - 6.29 |
| area | - 210.1 $ft^2$ |
| thickness | - 15 to 24%c |
| chord/diameter | - 0.2 |

In the second series, the baseline thickness ratio (21%) was used and chord was varied from 4 to 8 feet. The planform characteristics of these wings are shown in Figure 3-6.

The drag of the wings was estimated using the methods of Reference 7. The resulting drag increments for the composite wings are given in Figure 3-7. These values were added to the $f_e$ of the baseline aircraft.

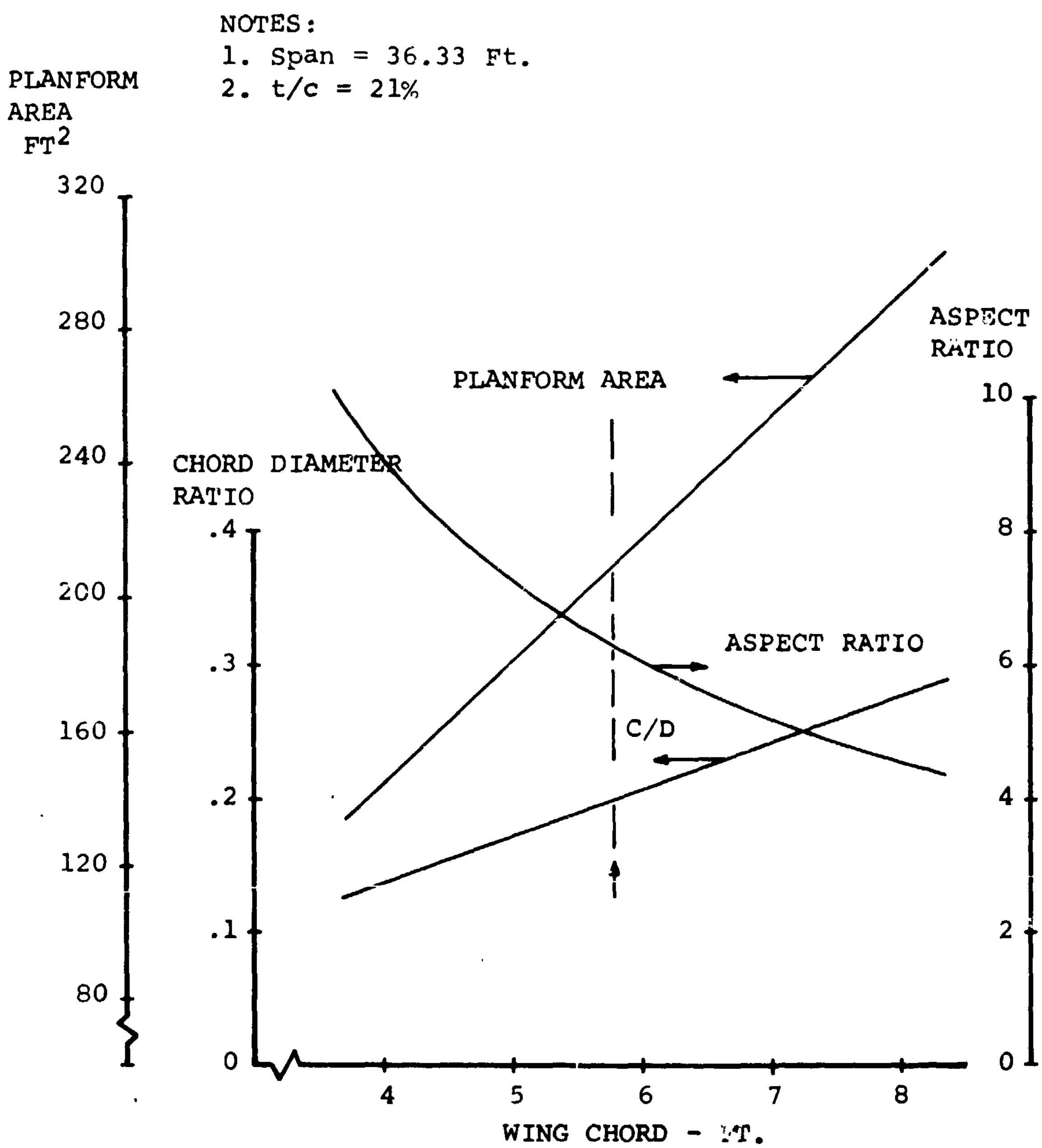



FIGURE 3-6: PLANFORM CHARACTERISTICS OF CONSTANT THICKNESS RATIO WING SERIES

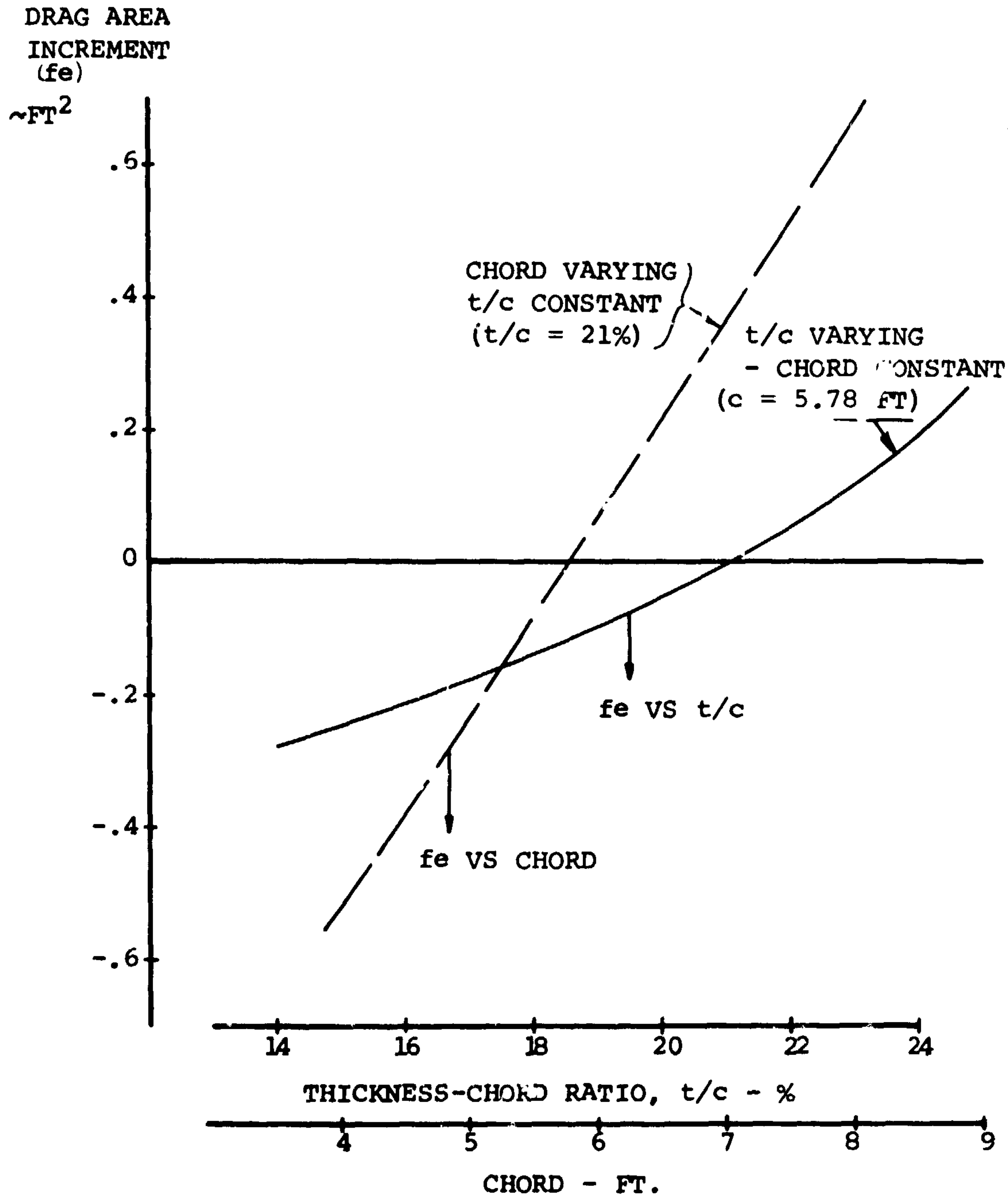



FIGURE 3-7: DRAG AREA INCREMENT BETWEEN COMPOSITE WINGS AND BASELINE ALL-METAL WING

A structural analysis was done to obtain the weights of the composite wings. This was done so that the weights would reflect the effects of wing design ground rules particularly with regard to strength and stiffness requirements (Section 3.2.5). The resulting wing weights are shown in Figure 3-8. It will be noted that the wings become stiffness critical at thicknesses below 17.15% and chords below 4.55 ft in the thickness and chord trades, respectively. At higher thickness and chord values the wings are strength critical.

The VASCOMP program (Reference 8) was used to compute the performance of the aircraft with the different wings. These results are presented in Figures 3-9 through 3-12.

The performance benefits obtainable with composites are expressed in two ways: in terms of the improvement in maximum rescue weight capability and in terms of the additional radius or midpoint hover time obtained at a given rescue capability. In the first case takeoff gross weight was reduced to take advantage of the reduced weight of the composite wing Fuel required was computed for the SAR mission profile (Section 3.2.1). (VASCOMP has a procedure that solves for TOGW when OWE and payload are given.) The reduced gross weight at the midpoint allowed an increase in the maximum rescue weight. These benefits are indicated by the curves labeled "Constant Mission Capability" in Figures 3-9 and 3-11.

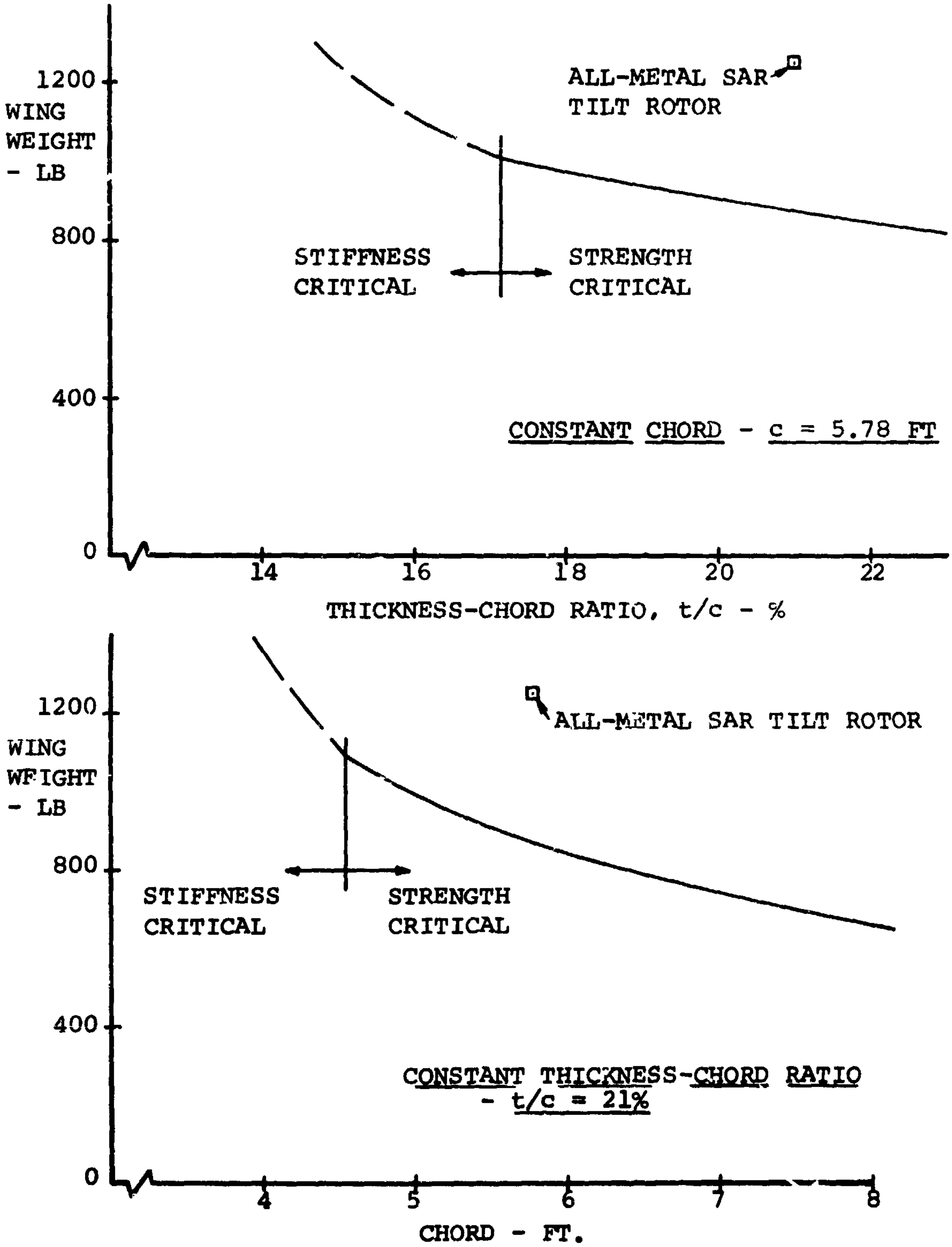



FIGURE 3-8: COMPOSITE WING WEIGHT FOR WING PARAMETER STUDY

In the second case takeoff gross weight was specified as 18025 lb. and the benefits due to the composite wings were put into an increased fuel load. This allowed either the mission radius to be increased over the basic 500 NM or the midpoint hover time to be increased over the basic 30 min. The maximum resuce capability in this case was maintained at 1400 lb or seven rescuees. The radius and hover time improvements are indicated by the curves labeled "Const TOGW" in Figures 3-10 and 3-12.

Also shown for reference are the all-metal and resized composite wing aircraft described in Sections 3.3 and 3.5, respectively, and the design point aircraft obtained by replacing the metal wing with a composite wing.

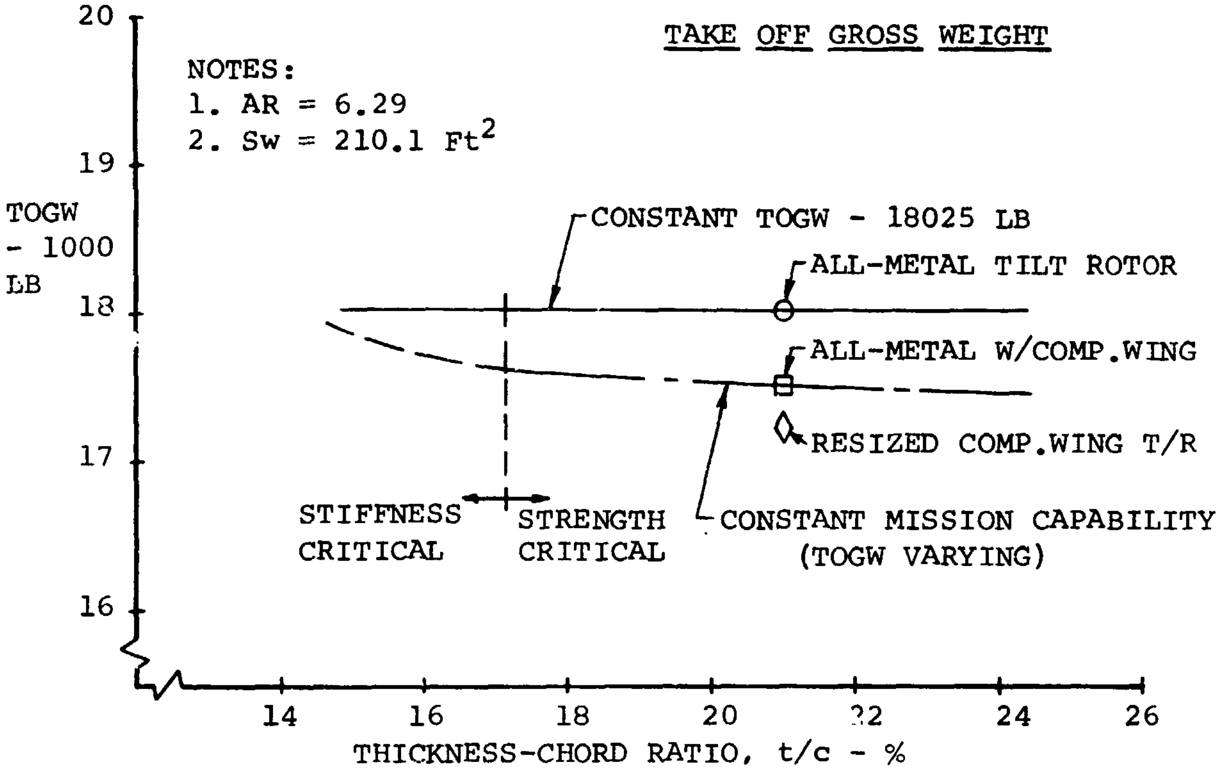
It will be noted in general that the curves reflect the strength critical - stiffness critical crossover resulting from the structural analysis. The benefits due to composites decrease rapidly as thickness or chord is decreased below the crossover because of the rapid increase in wing weight.

When constant mission capability is specified, replacing the metal wing with one of composite construction increases the rescue capability by more than 350 lb. With thicknesses greater than 19% or chords greater than 5.4 ft, two additional men can be picked up with a small margin in capability. The reduction in takeoff gross weight in these cases is about 500 lb.

When takeoff gross weight is fixed the composite wing will give mission radius increases ˜ 50 miles or more or midpoint hover time increases in the 15 to 20 minute range. Note that these cases are mutually exclusive. The additional fuel can be put into additional range or additional hover endurance but not both. Of course, both radius and endurance could be increased simultaneously, but not to the maxima shown.

The effect of composite construction in the wing is further illustrated by the design point aircraft shown in Figures 3-9 to 3-12. Replacing the metal wing with a composite wing reduced gross weight by about 500 lb. The reduced wing weight is reflected in empty weight and mission fuel. Resizing the aircraft with composites in the wing gave gross weight reduction of about 750 lb and decreased the rotor diameter from 28.9 ft to 27.1 ft. These comparisons are discussed in greater detail in Section 3.6.

The all-metal aircraft was used as a basis for the wing structural analysis presented in Section 4. The design gross weight used in the calculations shown reflects just the reduction in empty weight due to the reduction in wing weight. Maximum fuel for the all-metal aircraft was used in the structural analysis.



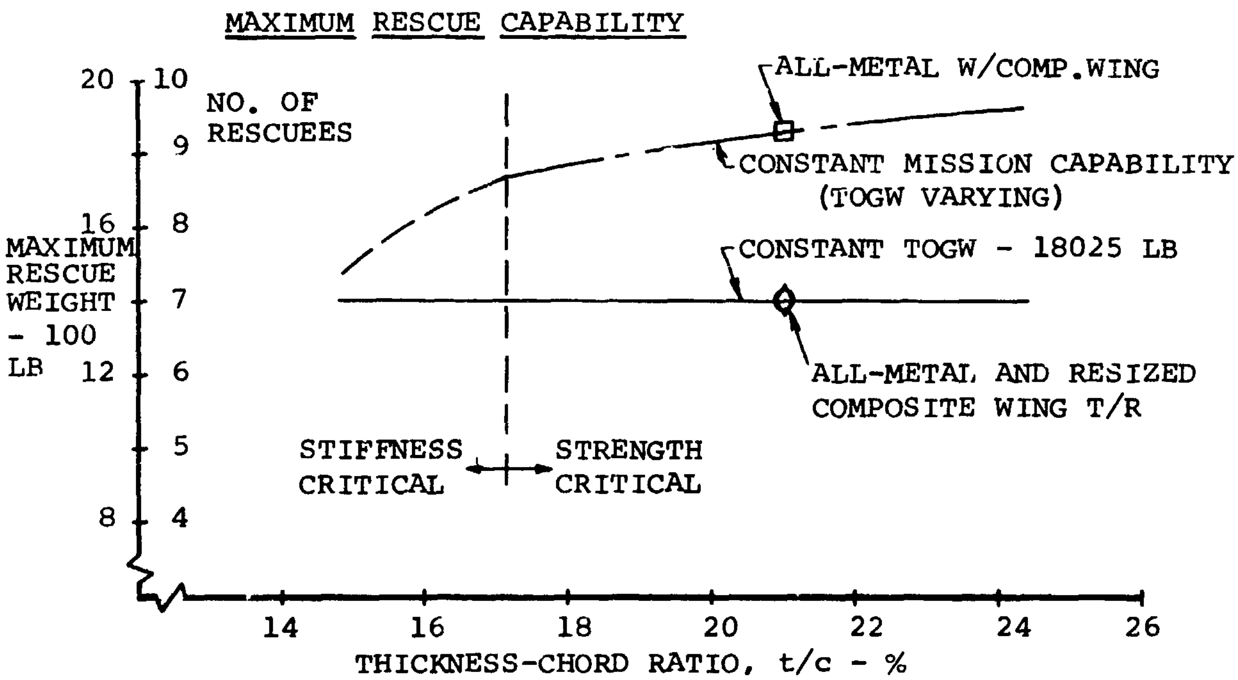




FIGURE 3-9: EFFECT OF WING THICKNESS RATIO (AT CONSTANT CHORD) ON TAKE OFF GROSS WEIGHT AND MAXIMUM RESCUE CAPABILITY

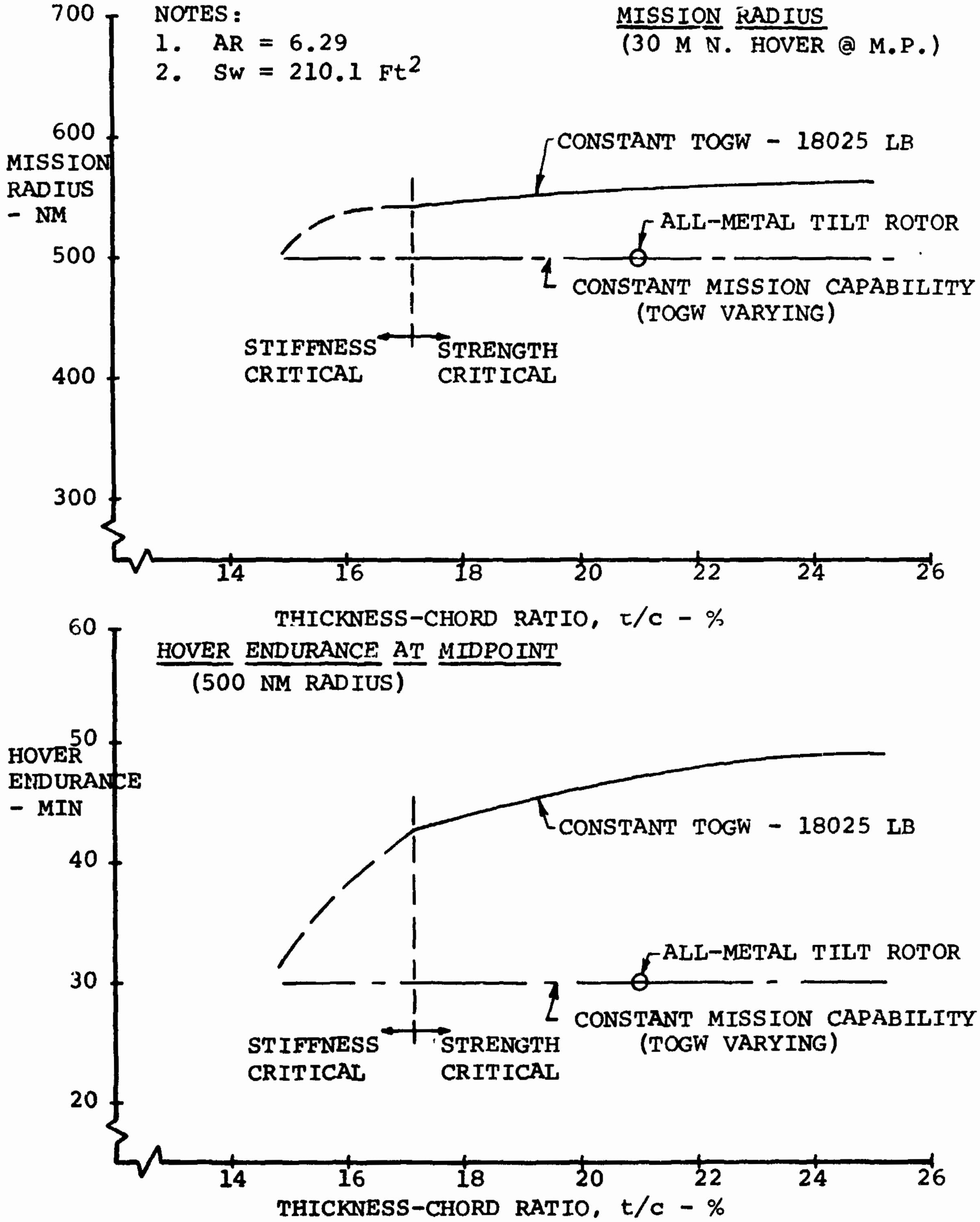



FIGURE 3-10: EFFECT OF WING THICKNESS RATIO (AT CONSTANT CHORD) ON MISSION RADIUS AND MIDPOINT HOVER ENDURANCE

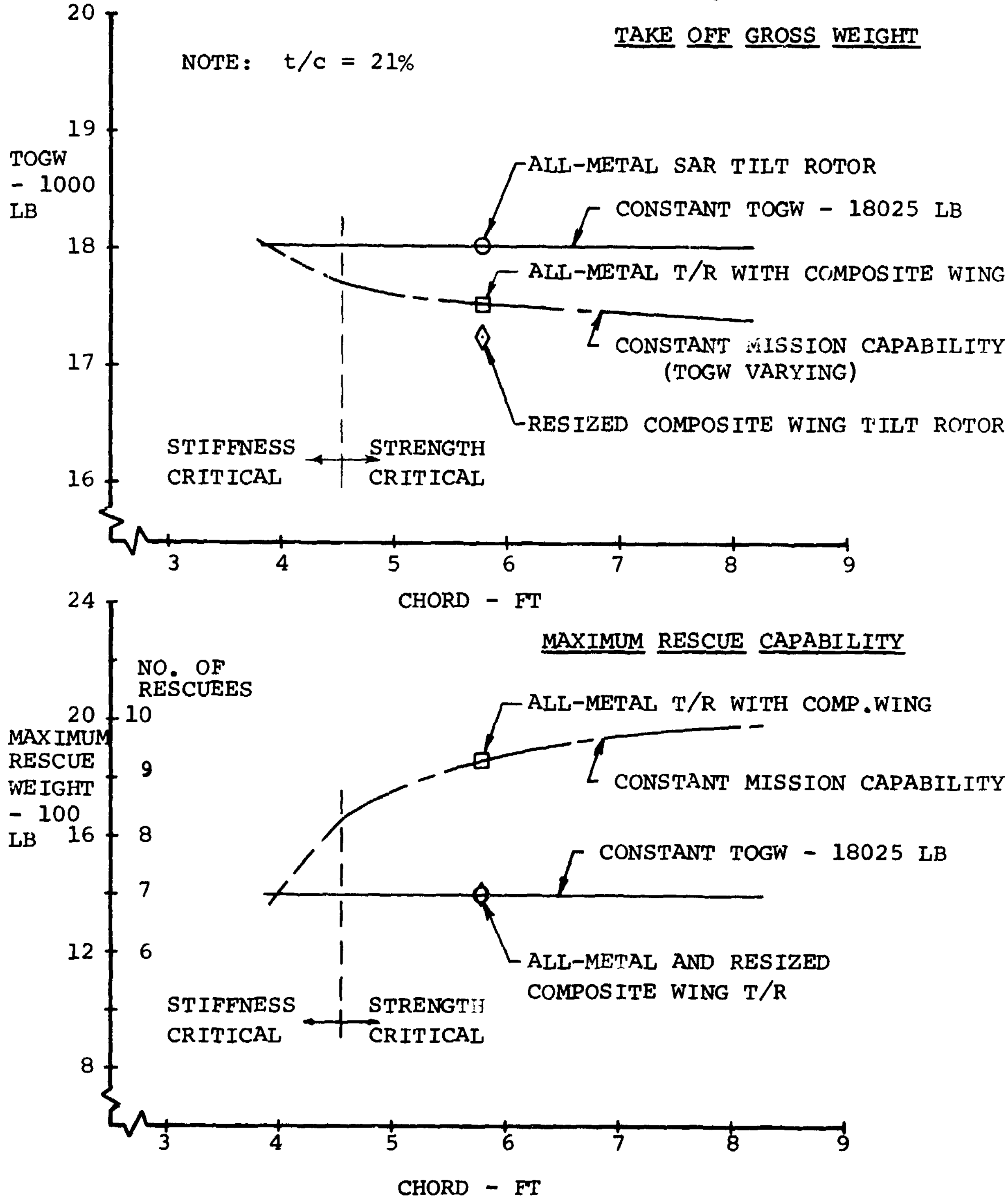



FIGURE 3-11: EFFECT OF WING CHORD (AT CONSTANT THICKNESS RATIO) ON TAKE OFF GROSS WEIGHT AND MAXIMUM RESCUE CAPABILITY

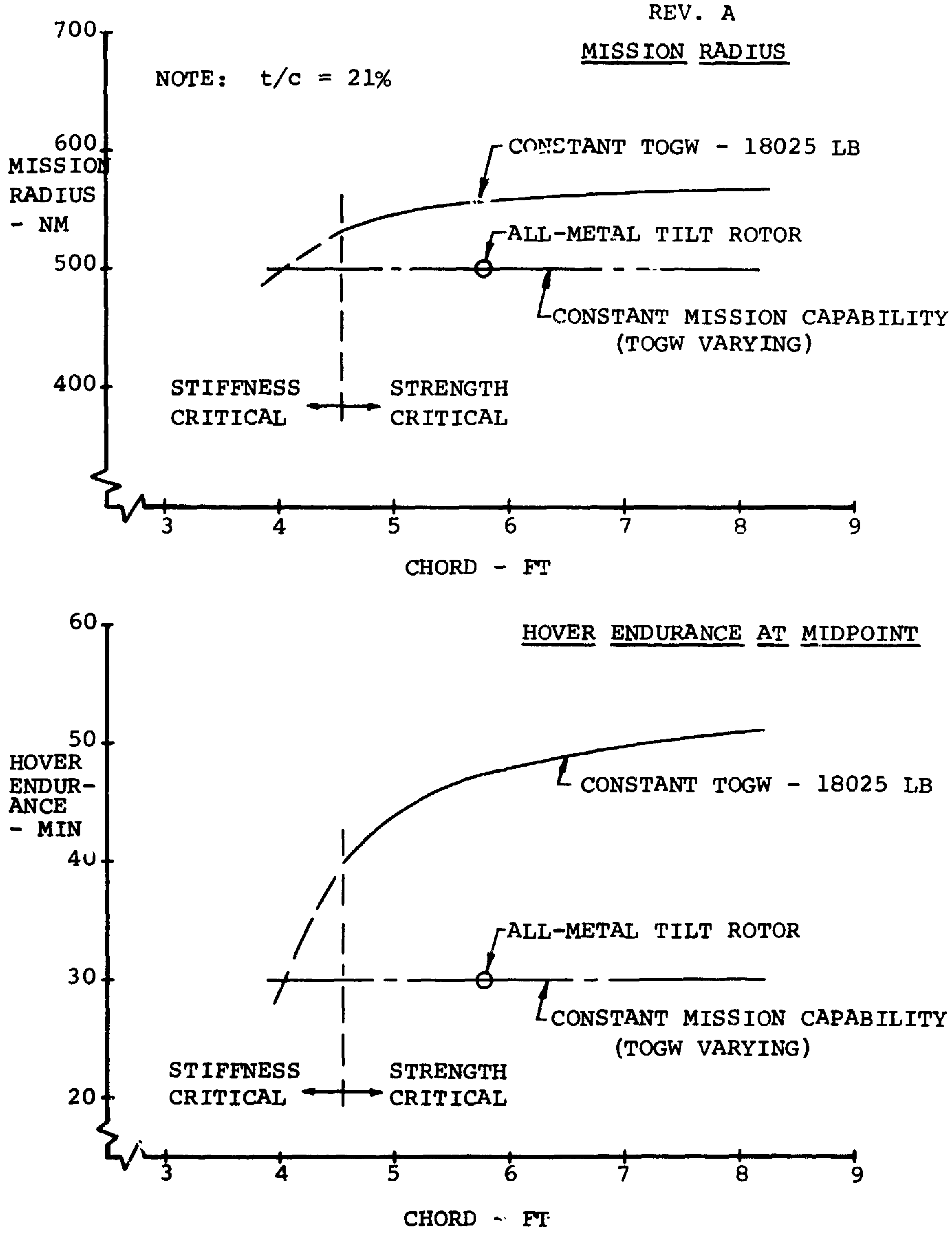



FIGURE 3-12: EFFECT OF WING CHORD (AT CONSTANT THICKNESS RATIO) ON MISSION RADIUS AND MIDPOINT HOVER ENDURANCE

## 3.5 RESIZED COMPOSITE WING AIRCRAFT

### 3.5.1 Description

Another way in which the benefits resulting from composite construction can be shown is in their effect on overall aircraft size and weight. To obtain these results the all-metal SAR tilt rotor (Section 3.3) was resized with the wing weight coefficient reduced by 30% to reflect composite construction. The 30% reduction factor has been agreed upon with USAF as being the weight saving obtainable with the use of composites (Section 3.2.6).

The parametric sizing results are shown in Figure 3-13. The data shown are: disk loading, midpoint thrust-weight ratio, and gross weight. In this instance the aircraft is sized by the disk loading limit (W/A=15 psf) and is just over the thrust-weight ratio requirement (T/W=1.1 with 7 rescuees). The aircraft therefore has nearly matched hover and cruise power requirements.

The composite wing tilt rotor aircraft has a design gross weight of 17242 pounds and 27.1 ft diameter rotors. The design characteristics of the aircraft are summarized as follows:

| | |
|---|---|
| Gross Weight (lb) | 17242 |
| Weight Empty (lb) | 11747 |
| Aspect Ratio | 6.37 |
| Wing Area ($ft^2$) | 187.7 |
| Wing Span (ft) | 34.55 |
| Wing Chord (ft) | 5.42 |

| | |
|---|---|
| Taper Ratio ($\lambda$)/Sweep Angle ($\Lambda$) | 1.0/0 |
| Wing Thickness | 21% |
| Wing Loading (psf) | 91.8 |
| Rotor Diameter (ft) | 27.1 |
| Chord to Diameter Ratio | 0.2 |
| Solidity Ratio | .087 |
| Disk Loading (psf) | 15.0 |
| Design $C_T/\sigma$ | .135 |
| Power Plant | (2) Lycoming PLT-27 |
| Rated Power @ SL/STD (SHP) | 1950 |

A summary weight statement for the aircraft is given in Table 3-3.

It is noted that this aircraft is very nearly the same size as the tilt rotor SAR aircraft described in Reference 1. That aircraft had a design gross weight of 16970 lb, an empty weight of 11500 lb, and a rotor diameter of 27 ft. Although the weight reduction factor used to account for composites is larger, the composite wing aircraft of this study is heavier because the fuselage and empennage weights do not include composites. In addition detail design studies have indicated that actual wing weights are greater than those indicated by the weight trends originally used. The wing weight trends used in this study therefore reflect the results of further studies in the design of the wing.

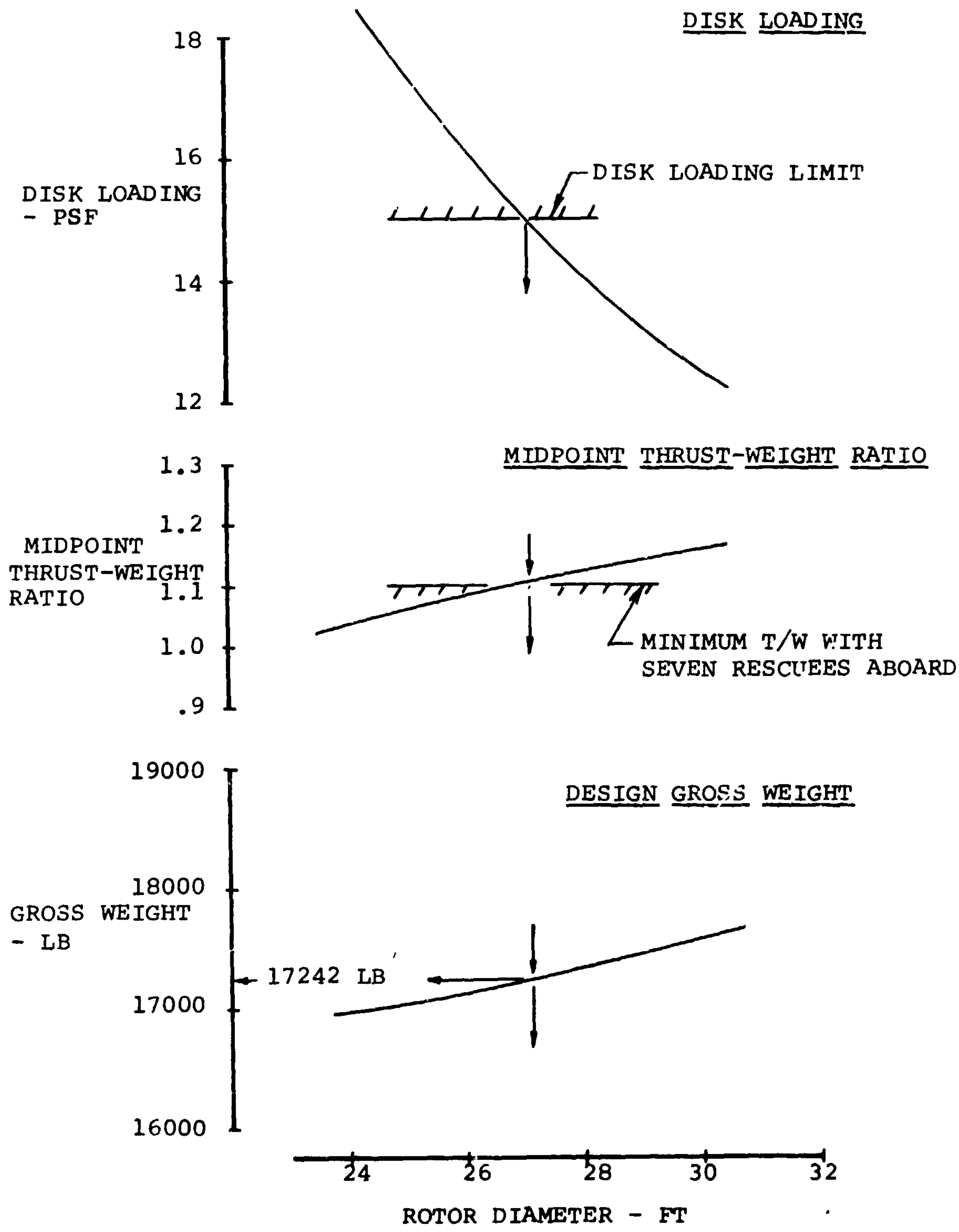



FIGURE 3-13: COMPOSITE WING SAR TILT ROTOR PARAMETRIC SIZING RESULTS

TABLE 3-3: SUMMARY WEIGHT STATEMENT - RESIZED COMPOSITE WING AIRCRAFT

| | | | | | | |
|---|---|---|---|---|---|---|
| ENG RATING @ SL/STD<br>ROTOR DIA/σ<br>WING AREA | 1950<br>27.1/.087<br>210.1 $FT^2$ | | | | | |
| ROTOR GROUP | 1164 | | | | | |
| WING GROUP | 812 | | | | | |
| TAIL GROUP | 230 | | | | | |
| BODY GROUP | 1450 | | | | | |
| BASIC | 2275 | | | | | |
| SECONDARY | | | | | | |
| SECOND.-DOORS, ETC. | | | | | | |
| ALIGHTING GEAR | 595 | | | | | |
| FLIGHT CONTROLS | 1341 | | | | | |
| ENGINE SECTION | 350 | | | | | |
| PROPULSION GROUP | | | | | | |
| ENGINES(S) | [illegible] | | | | | |
| AIR INDUCTION | | | | | | |
| EXHAUST SYSTEM | | | | | | |
| COOLING SYSTEM | 200 | | | | | |
| LUBRICATING SYSTEM | | | | | | |
| FUEL SYSTEM | 430 | | | | | |
| ENGINE CONTROLS | | | | | | |
| STARTING SYSTEM | | | | | | |
| PROPELLER INST. | | | | | | |
| *DRIVE SYSTEM | 1355 | | | | | |
| | | | | | | |
| AUX. POWER PLANT | - | | | | | |
| INSTR. AND NAV. | 135 | | | | | |
| HYDR. AND PNEU. | 130 | | | | | |
| ELECTRICAL GROUP | 800 | | | | | |
| ELECTRONICS GROUP | 1400 3200 | | | | | |
| ARMAMENT GROUP | 175 | | | | | |
| FURN. & EQUIP. GROUP | 350 | | | | | |
| PERSON. ACCOM. | | | | | | |
| MISC. EQUIPMENT | | | | | | |
| FURNISHINGS | | | | | | |
| EMERG. EQUIPMENT | | | | | | |
| AIR COND. & DE-ICING | 100 | | | | | |
| PHOTOGRAPHIC | | | | | | |
| AUXILIARY GEAR | 110 | | | | | |
| | | | | | | |
| | | | | | | |
| MFG. VARIATION | | | | | | |
| WEIGHT EMPTY | 11747 | | | | | |
| FIXED USEFUL LOAD | | | | | | |
| CREW (4) | 860 | | | | | |
| TRAPPED LIQUIDS | 40 | | | | | |
| ~~[illegible]~~ OWE | (12647) | | | | | |
| MISSION EQUIP. | 150 | | | | | |
| FUEL | 4300 | | | | | |
| CARGO | | | | | | |
| PASSENGERS/TROOPS | | | | | | |
| GUN & AMMO | 145 | | | | | |
| GROSS WEIGHT | 17242 | | | | | |

* INCLUDES ___ LB. XMNS. OIL

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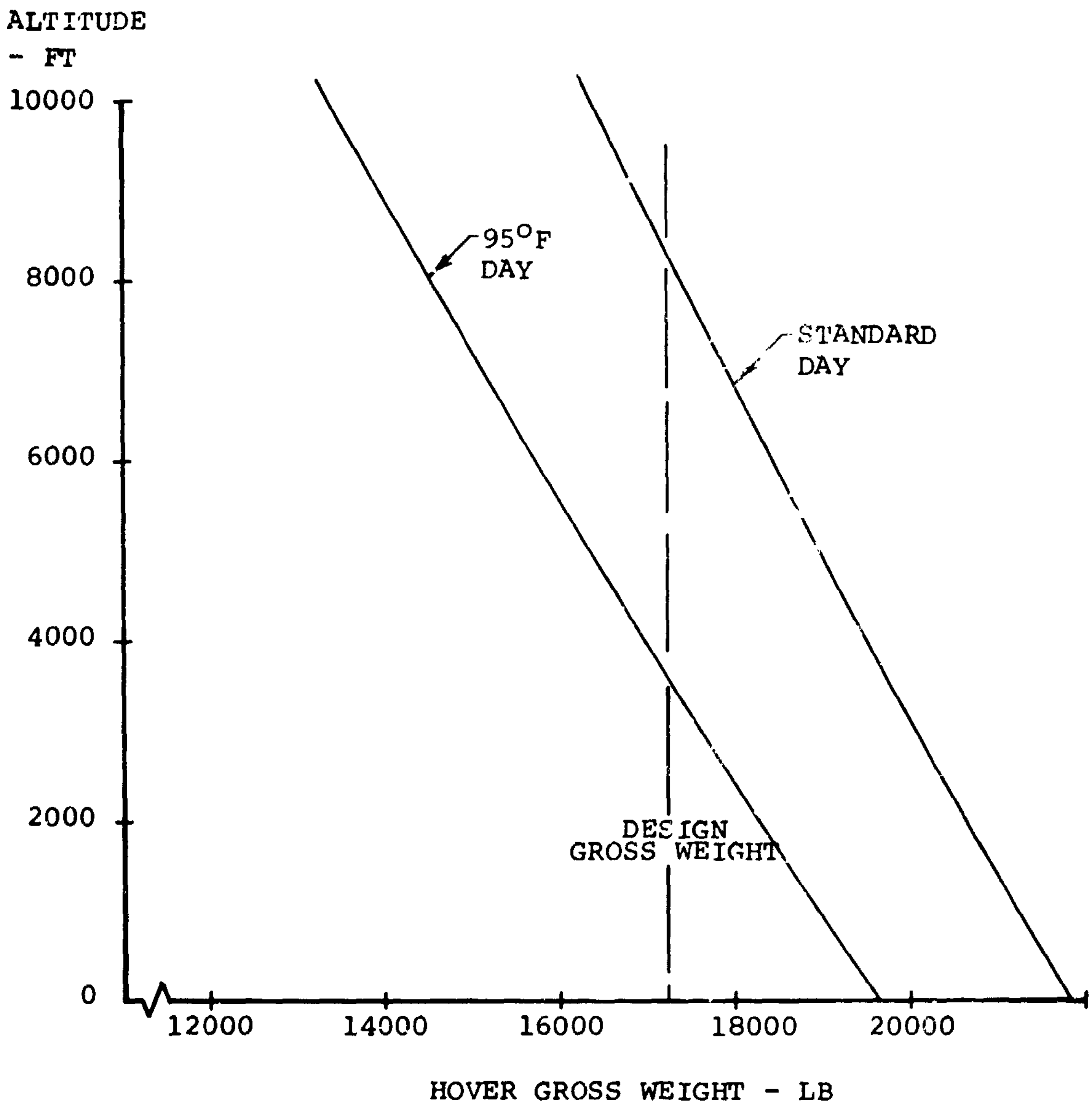
### 3.5.2 Performance

The performance characteristics of the resized composite wing aircraft are summarized in Figures 3-14 and 3-15. The data, presented for 17242 lb gross weight, include hover ceiling, flight envelope, and climb characteristics.

The aircraft can hover at design gross weight at 3600 ft on a 95°F day and at 8300 ft on a standard day (Figure 3-14). This performance is based on a thrust to weight ratio of 1.1 which allows 5% margin for download and 5% for maneuvarability.

Cruise mode performance is summarized in Figure 3-15. The aircraft has a sea level normal power speed of 326 kt. and can exceed 300 KTAS up to almost 19000 ft. Climb performance is good with absolute ceilings in excess of 25000 ft.




NOTES:
1. T/W = 1.1
2. Military Power
3. Rotor Tip Speed - 750 FPS
4. Design Gross Weight = 17242 LB

FIGURE 3-14: COMPOSITE WING SAR TILT ROTOR OGE HOVER CEILING

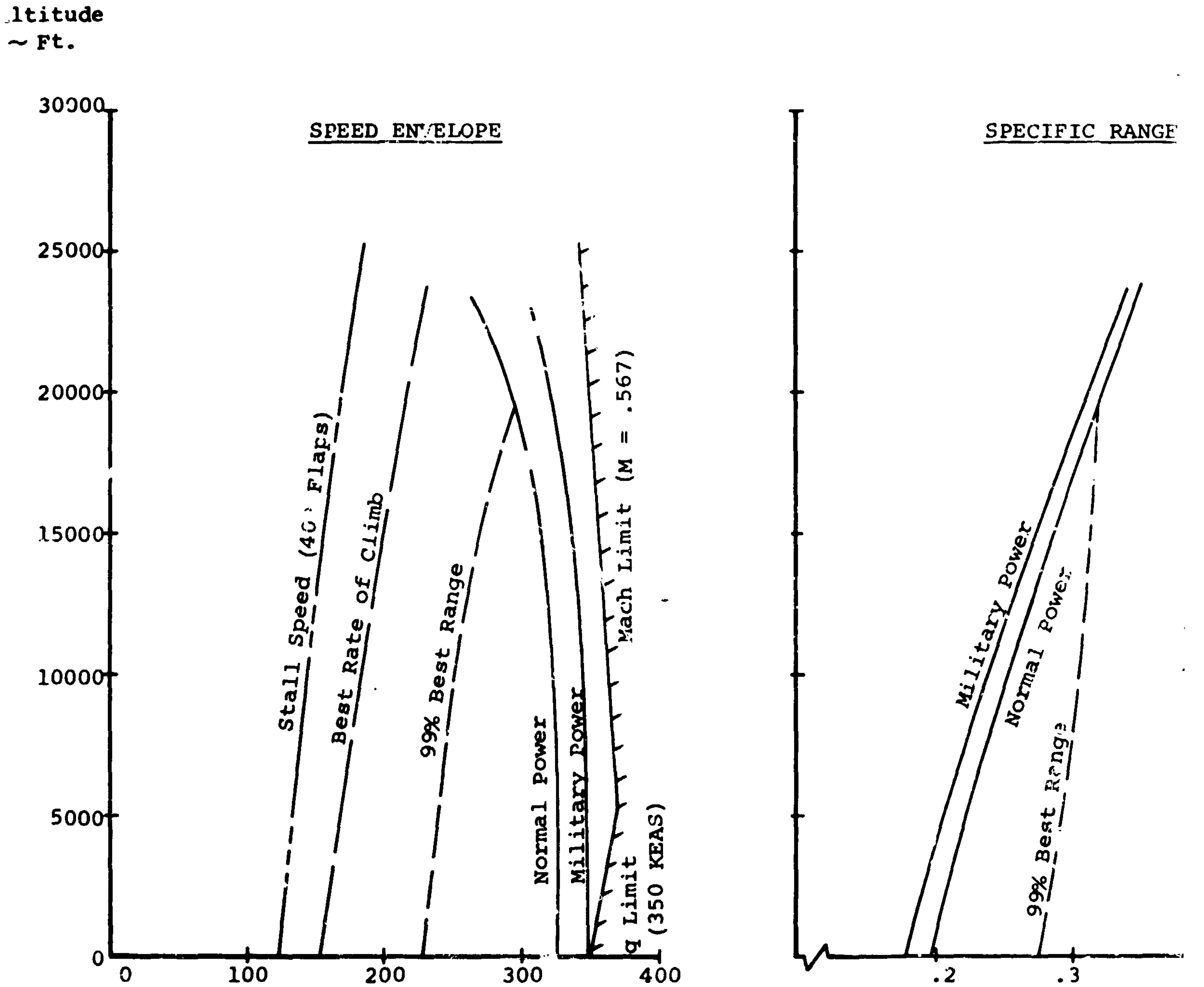
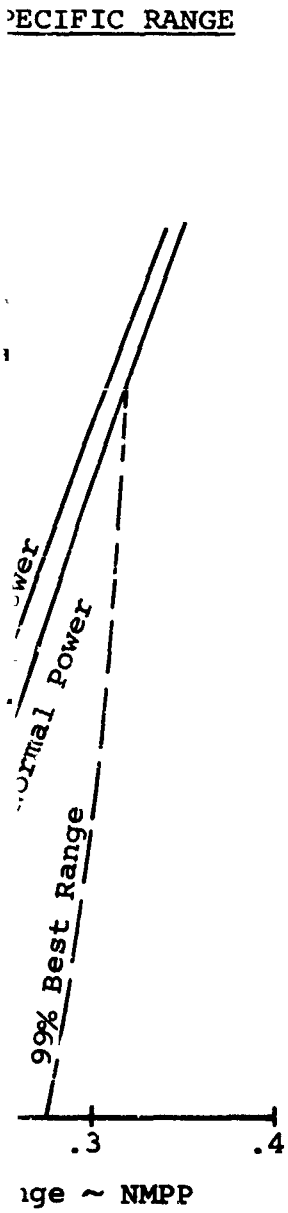
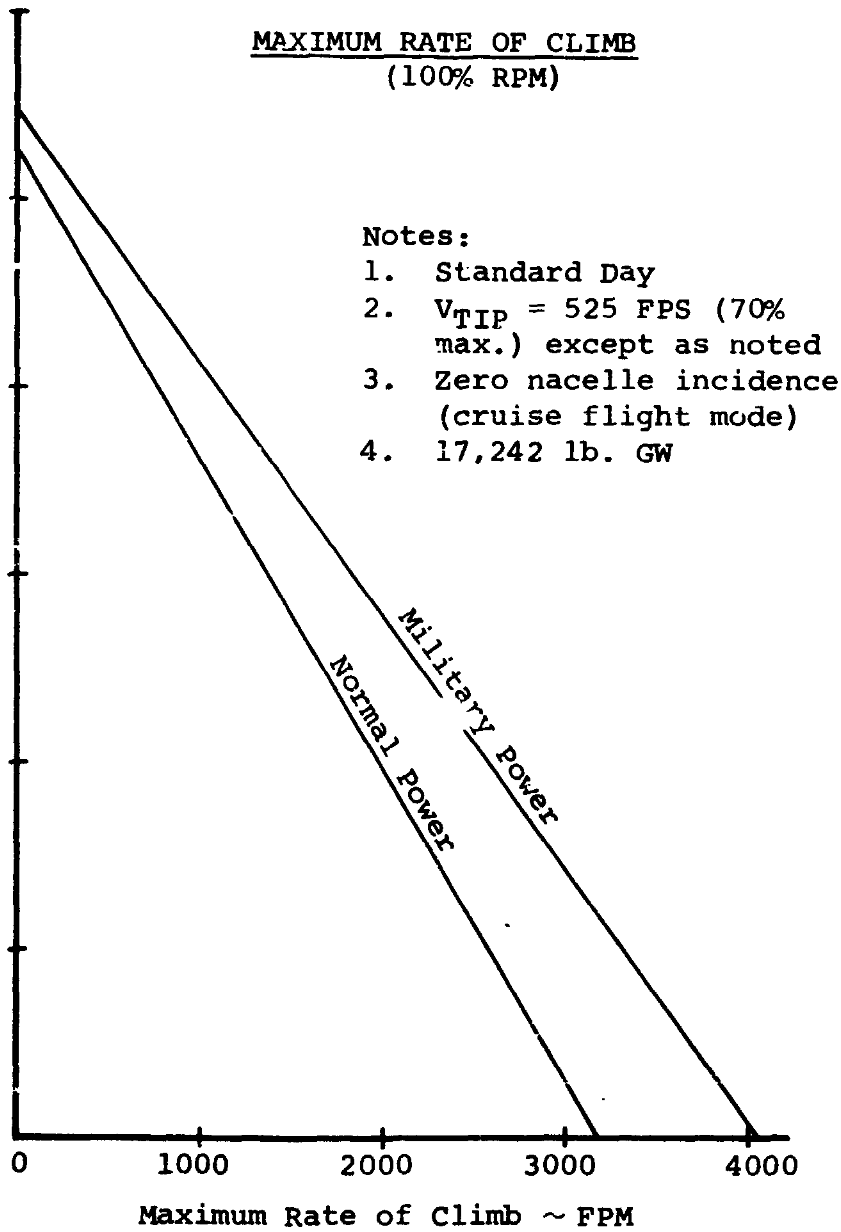



Figure 3-15. Composite Wing SAR Tilt R... Cru Performance Summary



OUT FRAME




ılt Rotor Cruisɩ

FOLDOUT FRAME 2

## 3.6 DESIGN BENEFITS OBTAINABLE WITH COMPOSITES

From a configuration design point of view the chief benefits resulting from the use of composites are the reductions in aircraft size and weight that can be obtained. From a structural design point of view composites offer superior corrosion resistance, greater fatigue strength and reduced notch sensitivity (hence greater damage tolerance). (These are discussed in greater detail in Section 4.3).

Three separate design point aircraft are shown in Figures 3-9 and 3-11. These are the all-metal aircraft, the all-metal aircraft with composite wing, and the resized composite wing aircraft. The first two are identical except for wing construction. The third has been resized to extend the wing weight benefits into other components of the aircraft (rotors, drive system, etc.). Physical and performance characteristics of the three are summarized for comparison in Table 3-4. A weight comparison is given in Table 3-5.

Replacing the metal wing of the all-metal aircraft with a composite wing gave a reduction in gross weight of 506 lb. Of this amount 375 lb is attributable to the wing and the rest to a reduction in fuel required. Resizing with composites reduced the physical size of the aircraft as well as its weight. Rotor diameter, for example, dropped to 27.1 ft from 28.9 ft for the all-metal aircraft. Wing area was reduced to 187.7 sq.

ft. The effects of the composite wing are also seen in the reductions in group weights down the line for the resized aircraft. The total reduction in empty weight between the all-metal and resized aircraft is 633 lb which is 5.1% of the all-metal value. The total reduction in gross weight is 783 lb or 4.3%. The total reduction in wing weight is 438 lb or 35% of the all-metal wing weight. This includes the inherent weight reduction due to composites and the effects of smaller size.

The effect of composite construction is also seen in the fraction of empty weight attributable to the wing. The metal wing is 10.1% of the empty weight while the composite wing is only 6.9% of it. This factor would help to offset the increased cost of composite construction.

TABLE 3-4. DESIGN POINT AIRCRAFT COMPARISON

| Physical Characteristics | All-Metal Aircraft | All-Metal w/Composite Wing | Resized Comp.Wing Aircraft |
|---|---|---|---|
| Gross Weight (lb) | 18025 | 17519 | 17242 |
| Empty Weight (lb) | 12380 | 12005 | 11747 |
| Wing Span (ft) | 36.35 | 36.35 | 34.55 |
| Wing Area (sq.ft.) | 210.1 | 210.1 | 187.7 |
| Wing Thickness | 21% | 21% | 21% |
| Wing Loading 9psf) | 85.8 | 83.4 | 91.8 |
| Rotor Diameter (ft) | 28.9 | 28.9 | 27.1 |
| Disk Loading (psf) | 13.8 | 13.4 | 15.0 |
| Max. Hover Gross Weight: (lb) | | | |
| SL/STD | 22870 | 22870 | 21850 |
| SL/95°F | 20520 | 20520 | 19660 |
| Forward Flight Performance (SL/STD, Design Gross Weight) | | | |
| Max. Speed (Mil Pwr)(kt) | 342 | 342 | 348 |
| Best Range Speed (kt) | 225 | 222 | 228 |
| Specific Range @ $V_{BR}$- (NMPP) | .271 | .279 | .276 |
| Max. Rate of Climb (fpm) | 4010 | 4120 | 4050 |

TABLE 3-5: DESIGN POINT AIRCRAFT WEIGHT COMPARISON

| DIA/σ | 28.9/.08 | 28.9/.08 | 27.1/.087 | | |
|---|---|---|---|---|---|
| | ALL-METAL BASELINE AIRCRAFT | ALL-METAL A/C W/COM-POSITE WING | RESIZED COMPOSITE WING A/C | | |
| ROTOR GROUP | 1203 | 1203 | 1164 | | |
| WING GROUP | 1250 | 875 | 812 | | |
| TAIL GROUP | | | 230 | | |
| BODY GROUP | | | 1450 | | |
| BASIC | 2300 | 2300 | 2275 | | |
| SECONDARY | | | | | |
| SECOND.-DOORS, ETC. | | | | | |
| ALIGHTING GEAR | | | 595 | | |
| FLIGHT CONTROLS | 1385 | 1385 | 1341 | | |
| ENGINE SECTION | 350 | 350 | 350 | | |
| PROPULSION GROUP | (2692) | (2692) | | | |
| ENGINES(S) | 620 | 620 | 620 | | |
| AIR INDUCTION | | | | | |
| EXHAUST SYSTEM | | | | | |
| COOLING SYSTEM | 200 | 200 | 200 | | |
| LUBRICATING SYSTEM | | | | | |
| FUEL SYSTEM | 445 | 445 | 430 | | |
| ENGINE CONTROLS | | | | | |
| STARTING SYSTEM | | | | | |
| PROPELLER INST. | | | | | |
| *DRIVE SYSTEM | 1427 | 1427 | 1355 | | |
| | | | | | |
| AUX. POWER PLANT | – | | | | |
| INSTR. AND NAV. | 135 | | | | |
| HYDR. AND PNEU. | 130 | | | | |
| ELECTRICAL GROUP | 800 | | | | |
| ELECTRONICS GROUP | 1400 | | | | |
| ARMAMENT GROUP | 175 | | | | |
| FURN. & EQUIP. GROUP | 350 3200 | 3200 | 3200 | | |
| PERSON. ACCOM. | | | | | |
| MISC. EQUIPMENT | | | | | |
| FURNISHINGS | | | | | |
| EMERG. EQUIPMENT | | | | | |
| AIR COND. & DE-ICING | 100 | | | | |
| PHOTOGRAPHIC | | | | | |
| AUXILIARY GEAR | 110 | | | | |
| | | | | | |
| | | | | | |
| MFG. VARIATION | | | | | |
| WEIGHT EMPTY | 12380 | 12005 | 11747 | | |
| FIXED USEFUL LOAD | | | | | |
| CREW (4) | 860 | 860 | 860 | | |
| TRAPPED LIQUIDS | 40 | 40 | 40 | | |
| ~~ENGINE OIL~~ OWE | (13280) | (12905) | (12647) | | |
| MISSION EQUIP. | 150 | 150 | 150 | | |
| FUEL | 4450 | 4319 | 4300 | | |
| CARGO | | | | | |
| PASSENGERS/TROOPS | | | | | |
| GUN & AMMO | 145 | 145 | 145 | | |
| GROSS WEIGHT | 18025 | 17519 | 17242 | | |

* INCLUDES ——— LB. XMNS. OIL

REV.

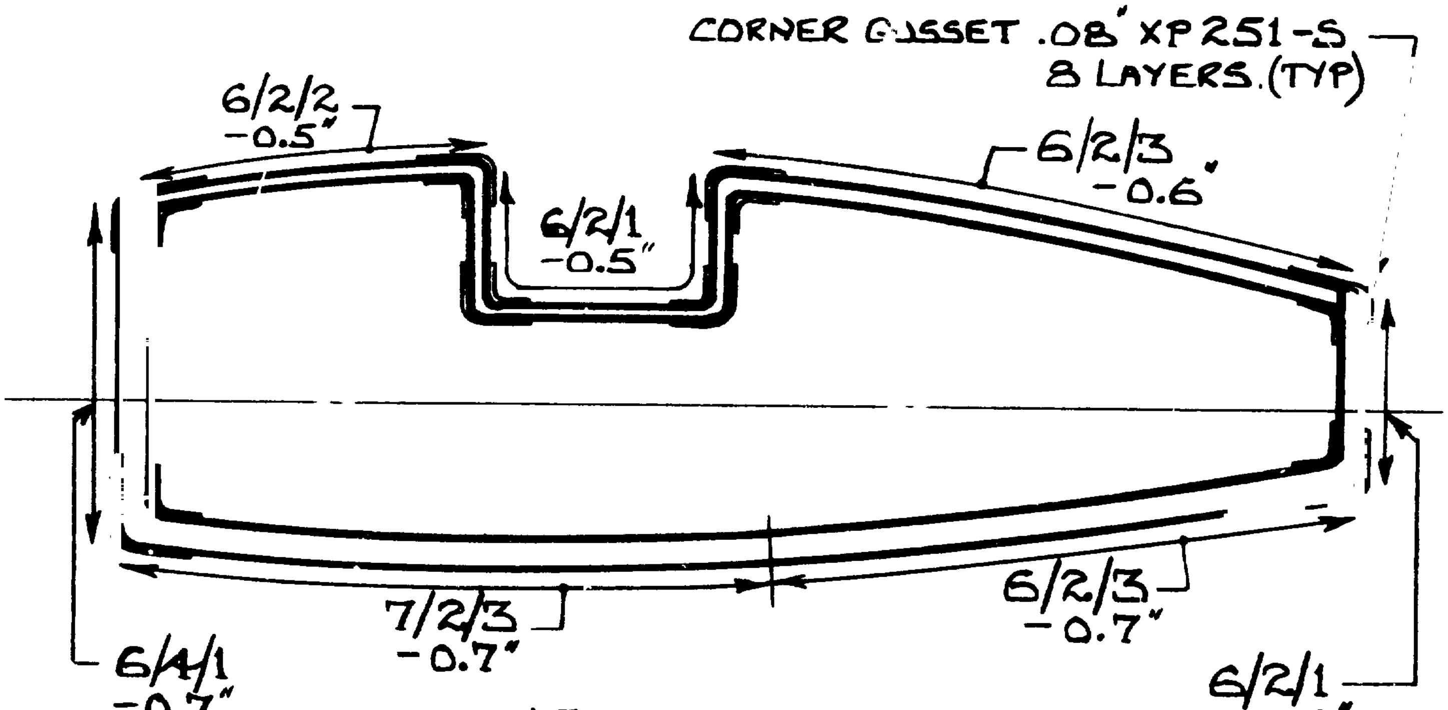
# 4.0 ADVANCED DESIGN STUDIES

This section presents the design and stress analysis of a composite wing torque box for a tilt rotor aircraft for the USAF-SAR role.

Two concepts for the wing torque box configuration were investigated, namely, a multi-spar and a two-spar torque box. For reasons discussed in Section 4.4, the two-spar configuration has been chosen for the torque box, as shown in Figures 4.1 and 4.2. A well in the upper surface provides space for cross shafting. The torque-box shell is a honeycomb sandwich with boron-epoxy facings on a fiberglass honeycomb core. All corners are gusseted using XP251-S glass cross ply to provide shear transfer capability and increase stability of the skin panels.

Although graphite-epoxy construction would result in a slight decrease in weight, boron epoxy was selected for this design for its superior impact resistance over graphite. This will provide the ruggedness required under normal service conditions and reduce maintenance costs.

The estimated weight is 626 lbs. for the boron torque box. Total weight for the composite wing is estimated at 875 lbs. An equivalent all-metal wing will weigh approximately 1250 lbs.




LEGEND

$\frac{X/X/X}{-Y} = \frac{N0°/N\pm45°}{-Y}/N90°$ WHERE N = Nº OF LAYERS.
Y = CORE DEPTH.

FIGURE 4-1. WING TORQUE BOX ROOT CONSTRUCTIONAL DETAILS AT WS 30

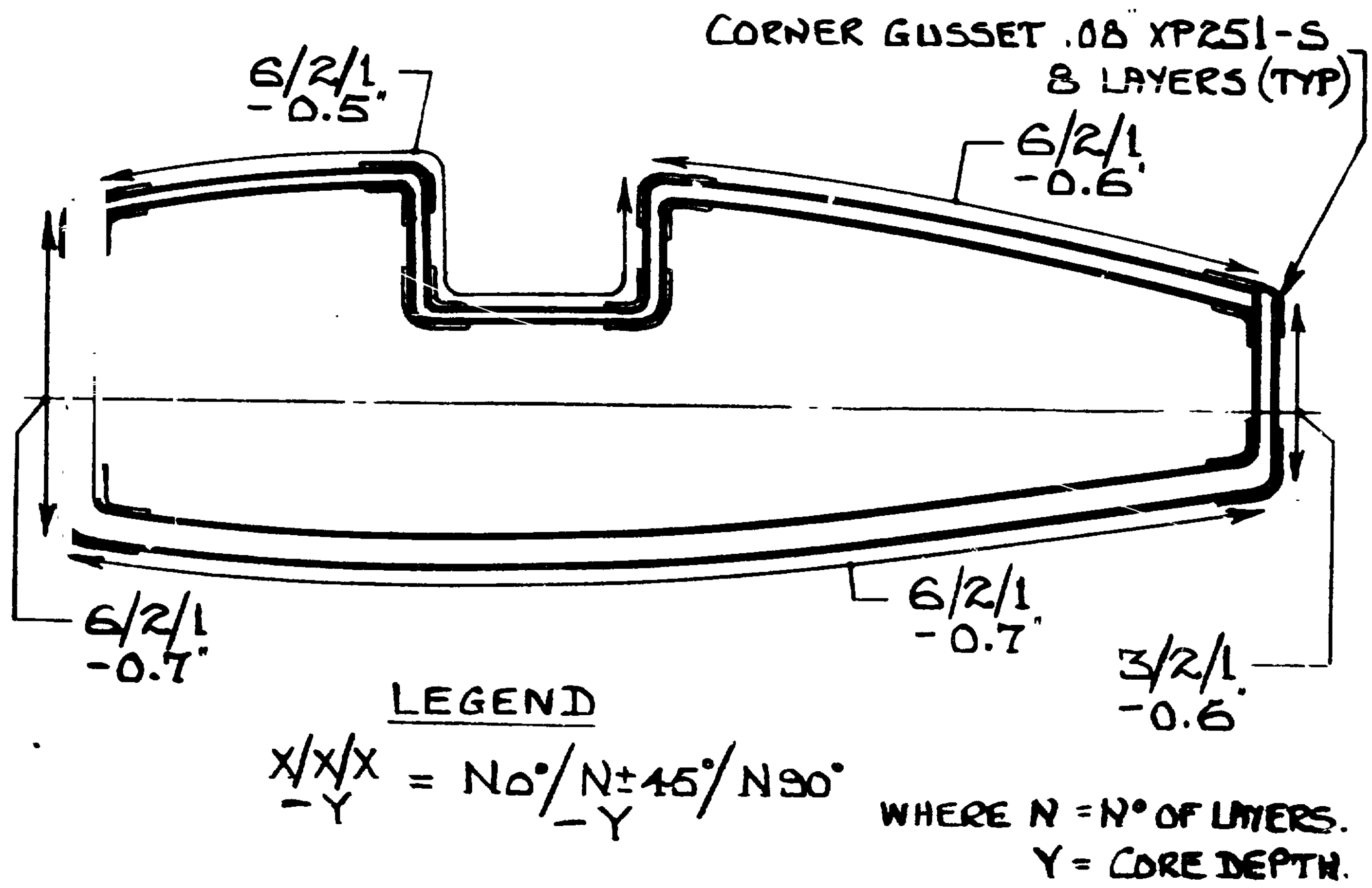



FIGURE 4-2. WING TORQUE BOX CONSTRUCTIONAL DETAILS AT WS 180

Thus, the composite design represents a weight saving of about 375 lbs. or 30% of the metal wing weight.

A summary of margins of safety is shown in Table 4-1.

TABLE 4-1. COMPOSITE WING TORQUE BOX - MARGINS OF SAFETY

| Element | Location (Wing Sta.) | Principal Load Condition | Margin of Safety |
|---|---|---|---|
| Upper Cover Forward | 30 | Compn. + Shear | 0.02 |
| | 180 | Shear | 0.33 |
| Upper Cover Aft | 30 | Compn. + Shear | 0.03 |
| | 180 | Compn. + Shear | 0.34 |
| Lower Cover Forward | 30 | Tension + Shear | .01 |
| | 180 | Tension + Shear | .18 |
| Lower Cover Aft | 30 | Compn. + Shear | .07 |
| | 180 | Compn. + Shear | .32 |
| Front Spar | 30 | Shear | .28 |
| | 180 | Shear | .23 |
| Rear Spar | 30 | Shear | .28 |
| | 180 | Shear | .5 |

## 4.1 STRUCTURAL DESIGN REQUIREMENTS

### 4.1.1 Basic Data

G. Wt. W = 17,650 Lbs.

Wing Span b = 36.3 Ft.

Wing Chord c = 5.78 Ft. (Constant)

Thickness Ratio = 0.21c

Front Spar at 0.15c

Rear Spar at 0.75c

Weight of Tilting and Fixed Nacelles = 2260 Lbs./Side

Ultimate Load Factor = 4.0g

Fuel (All in Wing) = 4450 Lbs.

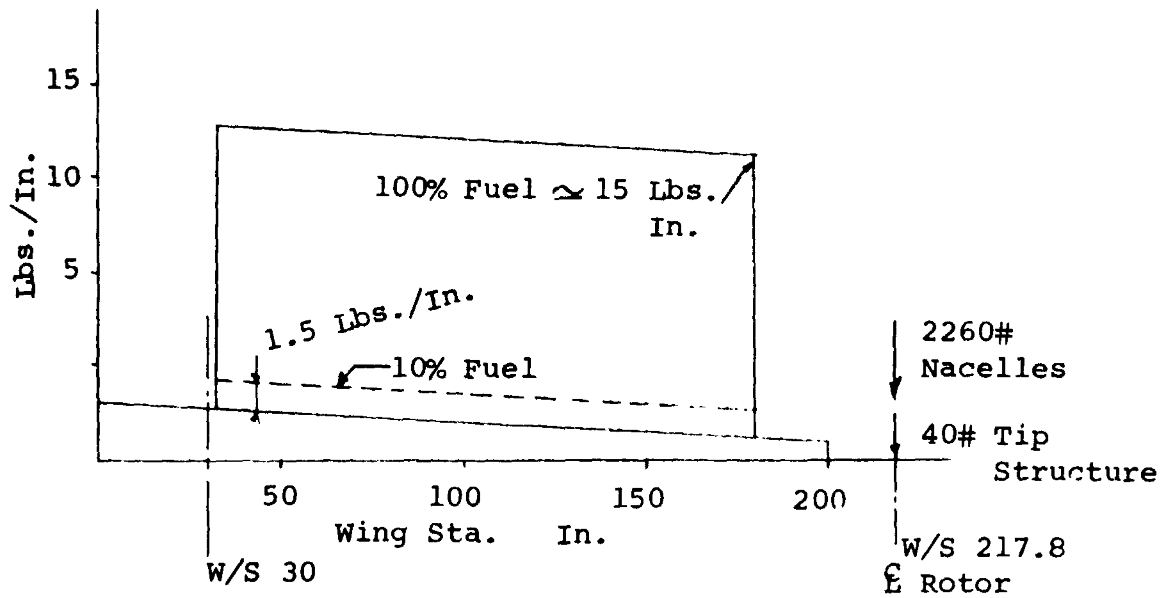
Wing Root Attachment at W.S. 30

### 4.1.2 Critical Design Condition

- Based on Model 222 structural analysis, design wing torque box to loads for flight condition 1 - VTO at 4g ultimate
- Check lower skin for compression loads during landing and ground taxi operations

## 4.2 LOAD DISTRIBUTION

### 4.2.1 Wing Mass Distribution




### 4.2.2 Rotor Loads 4.0g Condition (Ultimate)

a) 100% Fuel G.W. = 17,650 Lbs.

Rotor Download = 5% Rotor Thrust

$$\therefore \text{ Thrust } T = \frac{17650}{2} \times \frac{1}{.95} \times 4 = 37200 \text{ Lbs. (Ult.)}$$

b) 10% Fuel G.W. = 17650 -4450 +445

= 13645 Lbs.

$$\therefore \text{ Thrust } T = \frac{13645}{2} \times \frac{1}{.95} \times 4 = 28750 \text{ Lbs. (Ult.)}$$

Assume Wing Torsion = 500000 In.-Lbs. (Ult.)

#### 4.2.3 Spanwise BM and Shear Distribution

BM at W/S X, $30 \leq X \leq 180$, is given by

$$M_X = T\begin{pmatrix}217.8\\-X\end{pmatrix} - \left[2300\ (217.8-X) + \frac{(200-X)^2}{2} + \frac{1}{100}\frac{(200-X)^2}{2} \times \frac{(200-X)}{3}\right.$$

$$\left. + 15\ K\ \frac{(180-X)^2}{2}\right] \times 4\ -.05\ T\ \frac{(217.8-X)^2}{2 \times 217.8}$$

$$= (217.8-X)\left\{1-.000114(217.8-X)\right\} T - \left[9200(217.8-X)+2(200-X)^2\right.$$

$$\left. + \frac{1}{150}(200-X)^3 + 30K(180-X)^2\right]$$

$$V = T - 4\left[2300+(200-X)+\frac{1}{100}\frac{(200-X)^2}{2} + 15K(180-X)\right] - \frac{.05T}{217.8}(217.8-X)$$

$$= T - \left[10{,}000-4X+\frac{1}{50}(200-X)^2+60K(180-X)\right] -.000228(217.8-X)T$$

(K = Fraction of Fuel Remaining)

The bending moments and shears along the span are computed in Table 4-2 and shown graphically in Figures 4.3 and 4.4 respectively.

Table 4-2
USAF SAR Tilt Rotor Aircraft
Spanwise Bending Moment and Shear Di.
Cond. 1 (VTO) 4'g'

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | K = Fraction of Fuel Rema ning | | | 1.00 | | | |
| 2 | T = Rotor Thrust | | Lb. | 37200 | | | |
| 3 | X Wing Sta. | | In. | 180 | 150 | 100 | 50 |
| 4 | 217.8 - 3 | | In. | 37.8 | 67.8 | 117.8 | 167.8 |
| 5 | 1 - .000114 4 | Bending Moment | | .9957 | .9923 | .9866 | .9809 |
| 6 | 2 4 5 | | In. Lb. | 1400113 | 2502739 | 4323439 | 6122[illegible]35 |
| 7 | 9200 4 | | In. Lb. | 347760 | 623760 | 1083760 | 1543760 |
| 8 | 200 - 3 | | In. | 20 | 50 | 100 | 150 |
| 9 | 2 $8^2$ | | In. Lb. | 800 | 5000 | 20000 | 45000 |
| 10 | $8^3/150$ | | In. Lb. | 53 | 833 | 6667 | 22500 |
| 11 | 180 - 3 | | In. | 0 | 30 | 80 | 130 |
| 12 | 30 1 $11^2$ | | In. Lb. | 0 | 27000 | 192000 | 507000 |
| 13 | 7 + 9 + 10 + 12 | | In. Lb. | 348613 | 656593 | 1302427 | 2118260 |
| 14 | M = 6 - 13 | | In. Lb. | 1051500 | 1846146 | 3021012 | 4004675 |
| 15 | .000228 4 2 | Shear | Lb. | 322 | 578 | 1004 | 1420 |
| 16 | 4 3 | | Lb. | 720 | 600 | 400 | 200 |
| 17 | $8^2/50$ | | Lb. | 8 | 50 | 200 | 450 |
| 18 | 60 1 11 | | Lb. | 0 | 1800 | 4800 | 7800 |
| 19 | 10000 - 16 + 17 + 18 | | Lb. | 9288 | 11250 | 14600 | 18050 |
| 20 | V = 2 - 15 - 19 | | Lb. | 27590 | 24946 | 21596 | 17721 |

Shear at ℄ Rotor = 28000 Lbs. 100% Fuel
[illegible]550 Lbs. 10% Fuel



OLDOUT FRAME

Aircraft
Shear Distribution
^'g'

| | | | | | | |
|---|---|---|---|---|---|---|
| | | .1 | | | | |
| | | 28750 | | | | |
| 50 | 30 | 180 | 150 | 100 | 50 | 30 |
| 167.8 | 187.8 | 37.8 | 67.8 | 117.8 | 167.8 | 187.8 |
| .9809 | .9786 | .9957 | .9923 | .9866 | .9809 | .9706 |
| 6122935 | 6836556 | 1082077 | 1934241 | 3341368 | 4732107 | 5240512 |
| 1543760 | 1727760 | 347760 | 623760 | 1083760 | 1543760 | 1727760 |
| 150 | 170 | 20 | 50 | 100 | 150 | 170 |
| 45000 | 57800 | 800 | 5000 | 20000 | 45000 | 57800 |
| 22500 | 32753 | 53 | 833 | 6667 | 22500 | 32753 |
| 130 | 150 | 0 | 30 | 80 | 130 | 150 |
| 507000 | 675000 | 0 | 2700 | 19200 | 50700 | 67500 |
| 2118260 | 2493313 | 346813 | 632293 | 1129447 | 1661960 | 1885813 |
| 4004675 | 4343343 | 735264 | 1301948 | 2211921 | 3070147 | 3354699 |
| 1429 | 1600 | 249 | 446 | 776 | 1105 | 1236 |
| 200 | 120 | 720 | 600 | 400 | 200 | 120 |
| 450 | 578 | 8 | 50 | 200 | 450 | 578 |
| 7800 | 9000 | 0 | 180 | 480 | 780 | 900 |
| 18050 | 19458 | 9288 | 9630 | 10280 | 11030 | 11358 |
| 17721 | 16142 | 19213 | 18871 | 17694 | 16615 | 16156 |

FOLDOUT FRAME 2

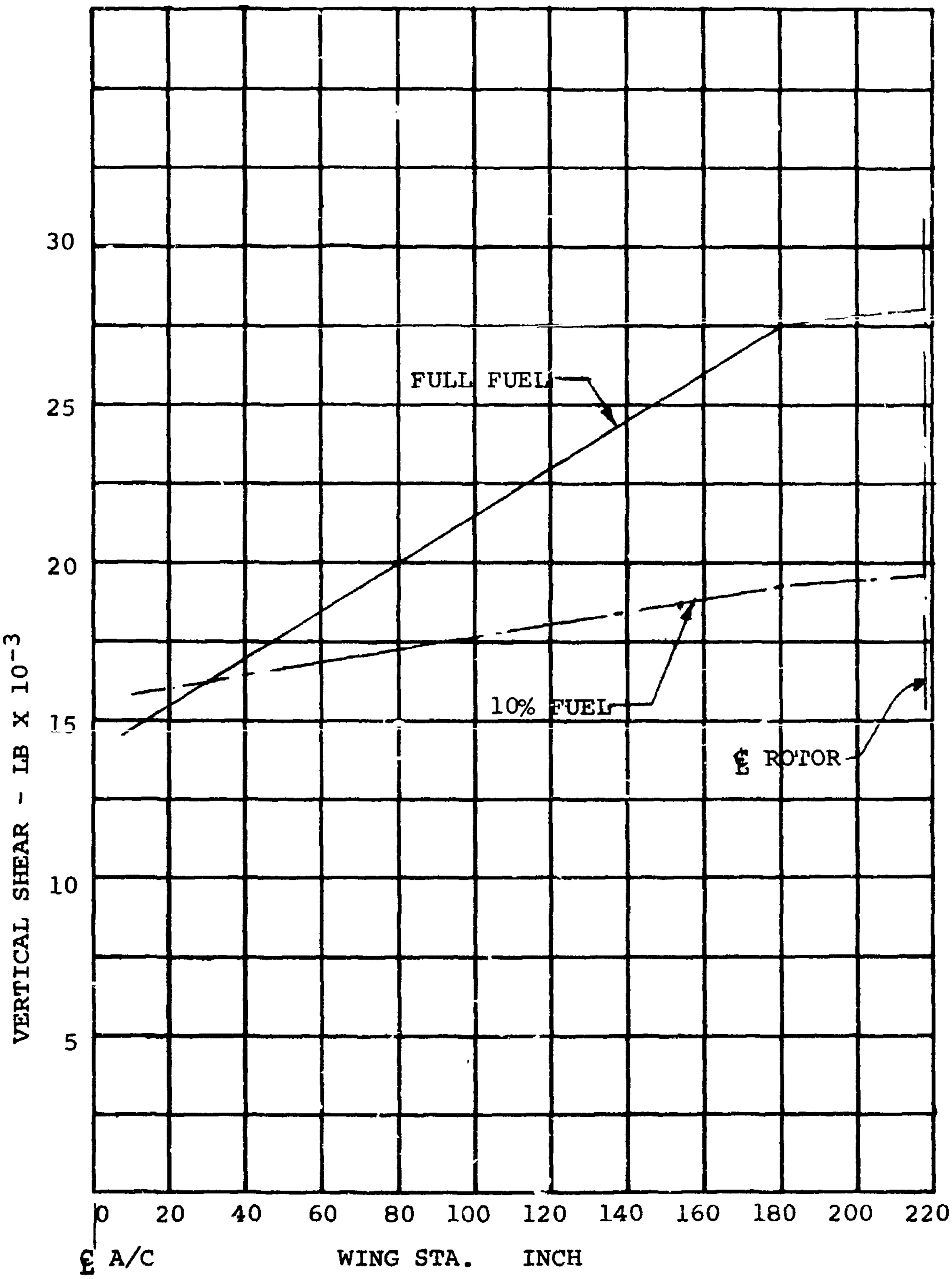



FIGURE 4-3. SPANWISE SHEAR DISTRIBUTION
COND. 1, 4'g' VTO

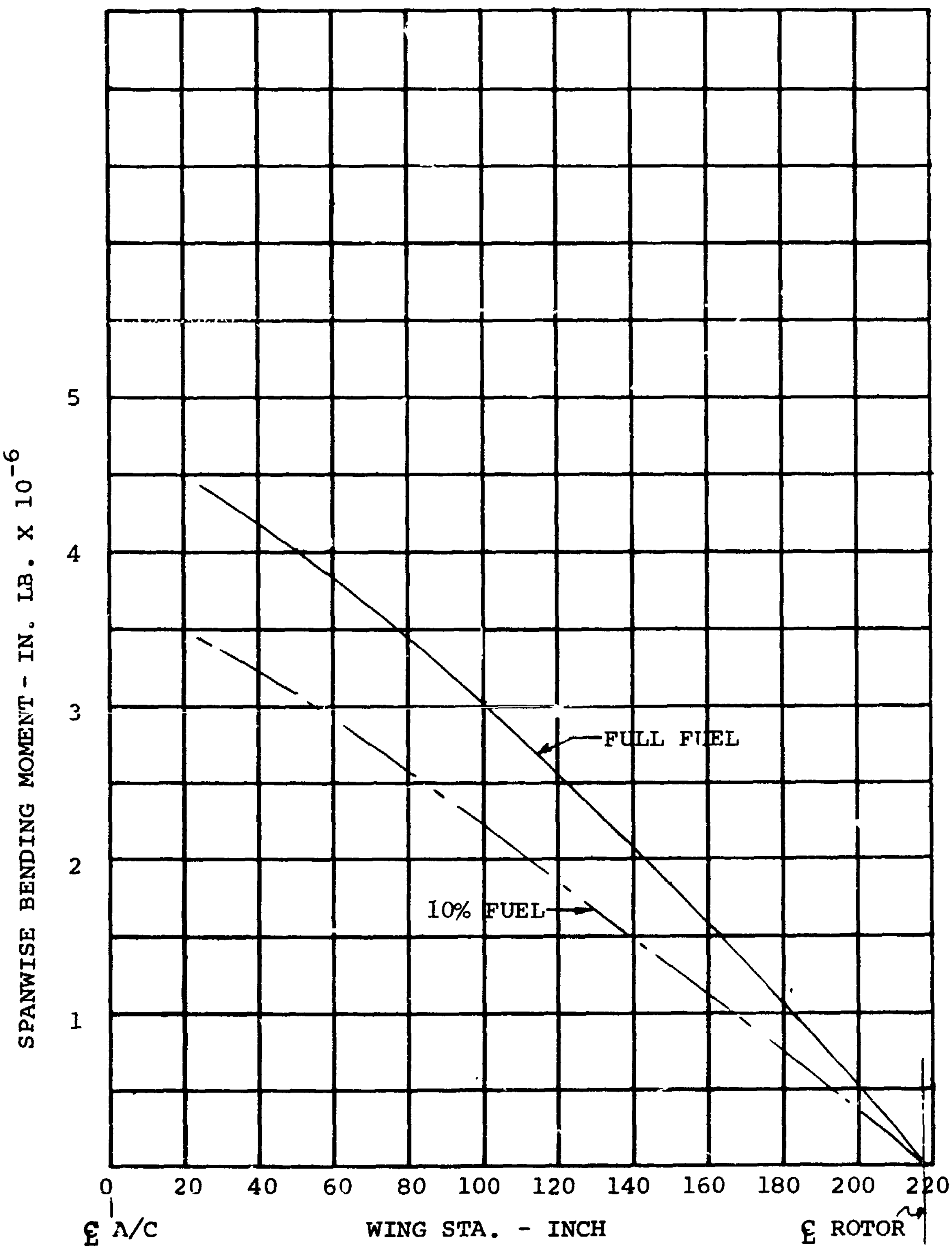



FIGURE 4.4. SPANWISE BM DISTRIBUTION
COND. 1, 4'g' VTO

### 4.2.4 Estimate of Stiffness Requirements

It is assumed that the stiffness distribution should match Model 222 wing stiffness and that the wing frequencies should be the same as for Model 222 wing designed for an ultimate LF = 4.0'g'

$$\frac{\text{SAR Wing Bending Stiffness}}{\text{Model 222 Wing Bending Stiffness}} = \frac{\left(2260+\frac{438}{4.2}\right)(18.15)^3}{.971^2 \times 10.32 \times 10^6}$$

$$= 1.45$$

i.e., $\frac{(EI)_{SAR}}{(EI)_{222}} = 1.45$

and $\frac{\text{SAR Wing Torsional Stiffness}}{\text{Model 222 Wing Torsional Stiffness}}$

$$= \frac{\left[(3.47 + .971)/3.47 \times .97\right]^2 \times 18.15}{30.6}$$

$$= 1.03$$

i.e., $\frac{GJ_{SAR}}{GJ_{222}} = 1.03$

The resulting EI and GJ distribution are shown in Figure 4.5.

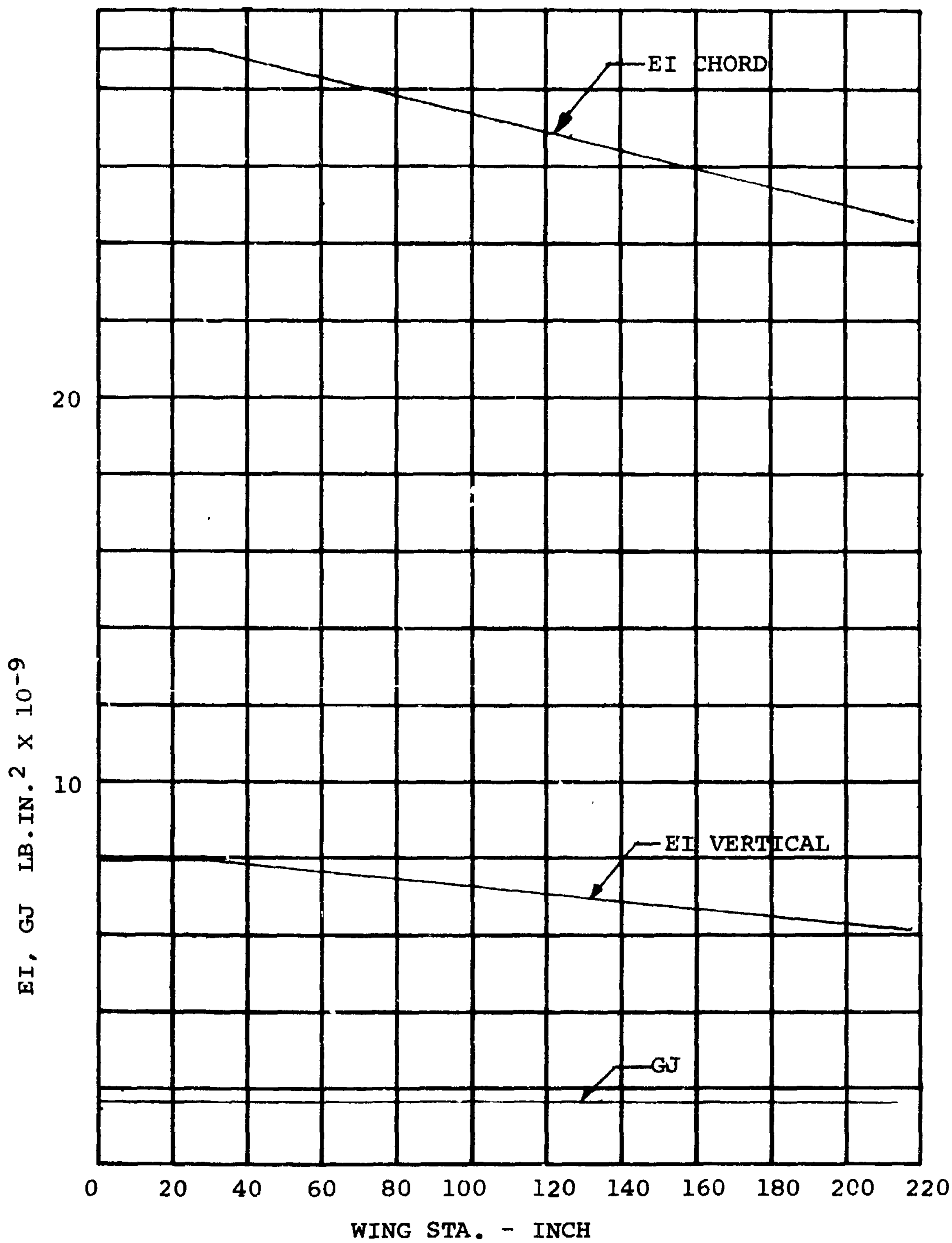



FIGURE 4_5. REQUIRED WING STIFFNESS DISTRIBUTION

## 4.3 MATERIAL SELECTION

### 4.3.1 FILAMENT-MATRIX SYSTEMS

The selection of the basic composite fiber has a major impact on the overall cost and performance of the system. Four basic filament-matrix systems (two of which are state-of-the-art and the others considered advanced) were evaluated for application in the wing structure.

- o Boron/Epoxy
- o S-Glass/Epoxy
- o E-Glass/Epoxy
- o Graphite/Epoxy

Representative values of the basic material properties are presented in Table 4-3. The values shown are design allowables, statistically reduced, based on component fatigue experience and extensive coupon testing. For graphite epoxies, considerable data are currently being generated in support of the HLH program.

Design allowables for composite materials are based on tests conducted under Army and Air Force sponsorship, as well as Boeing research. The design properties are derived from over 1,000 tests of boron/epoxy composites, 3,000 tests of glass/epoxy composites, and 350 tests of mixtures of glass and high-modulus composites. The data include effects of notches, temperature, humidity, load sequencing, effect of mean load,

TABLE 4-3. PRELIMINARY DESIGN ALLOWABLES

| Material | | 1002-S Glass | XP-251-S Glass | Boron/ Epoxy | Gra-phite/ Epoxy HT | Gra-phite/ Epoxy HM |
|---|---|---|---|---|---|---|
| $F_{tu}$ (ksi) | 0° | 175 | 247 | 178 | 143 | 95 |
| | ± 45° | 28.2 | 22.6 | 22 | 10 | |
| | 90° | 2.98 | 1.78 | 10 | 7.5 | |
| $F_{cu}$ (ksi) | 0° | 126 | 160 | 300 | 140 | 90 |
| | ± 45° | 31 | 24 | 30 | 9 | |
| | 90° | | 22 | 25 | 25 | 25 |
| $F_{su}$ (ksi) | 0° | 7.1 | 11.2 | 9 | 8.5 | 8 |
| | ± 45° | 27 | 48 | 67 | 35 | 24 |
| | 90° | | | | | |
| E (psi x $10^{-6}$) | 0° | 7.15 | 7.45 | 30 | 21.2 | 30.1 |
| | ± 45° | 1.8 | 2.0 | 3 | 2 | |
| | 90° | 1.74 | 2.99 | | 2 | |
| G (psi x $10^{-6}$) | 0° | .59 | 1.1 | 1 | .7 | |
| | ± 45° | 1.85 | 2.4 | 8.8 | 4 | 5 |
| | 90° | .55 | 1.1 | 1 | | |
| $\rho$ (Lb/In$^3$) | | .066 | .073 | .075 | .054 | .058 |
| Thickness Per Cured Lap (In.) | | .010 | .010 | .007 | .010 | .010 |

and failure modes. Although data regarding material properties are available for the most part, there are gaps which limit the application of these filament-matrix systems.

The above four basic filament-matrix systems were selected because of their range of cost, strength, stiffness, established performance confidence, and related experience existing within the Boeing Vertol. E-glass and S-glass have been used for several years and their basic properties are generally well-known.

Since most of the composite materials available are nonmetallic, their susceptibility to corrosion as it is commonly understood is negligible, (Galvanic corrosion should be considered when certain composites are in contact with metals. Especially susceptible is an aluminum/graphite interface). Currently available epoxy matrix systems are also highly resistant to environmental effects.

The plot of extensional modulus divided by density ($E/\rho$) and torsional modulus divided by density ($G/\rho$) in Figure 4.6 is an indication of the flexibility available to the designer in achieving a match of dynamic characteristics required for dynamic-critical wings while at the same time achieving weight savings.

Much of the primary structure of a typical metal V/STOL airplane is designed by fatigue considerations. The high ratio of

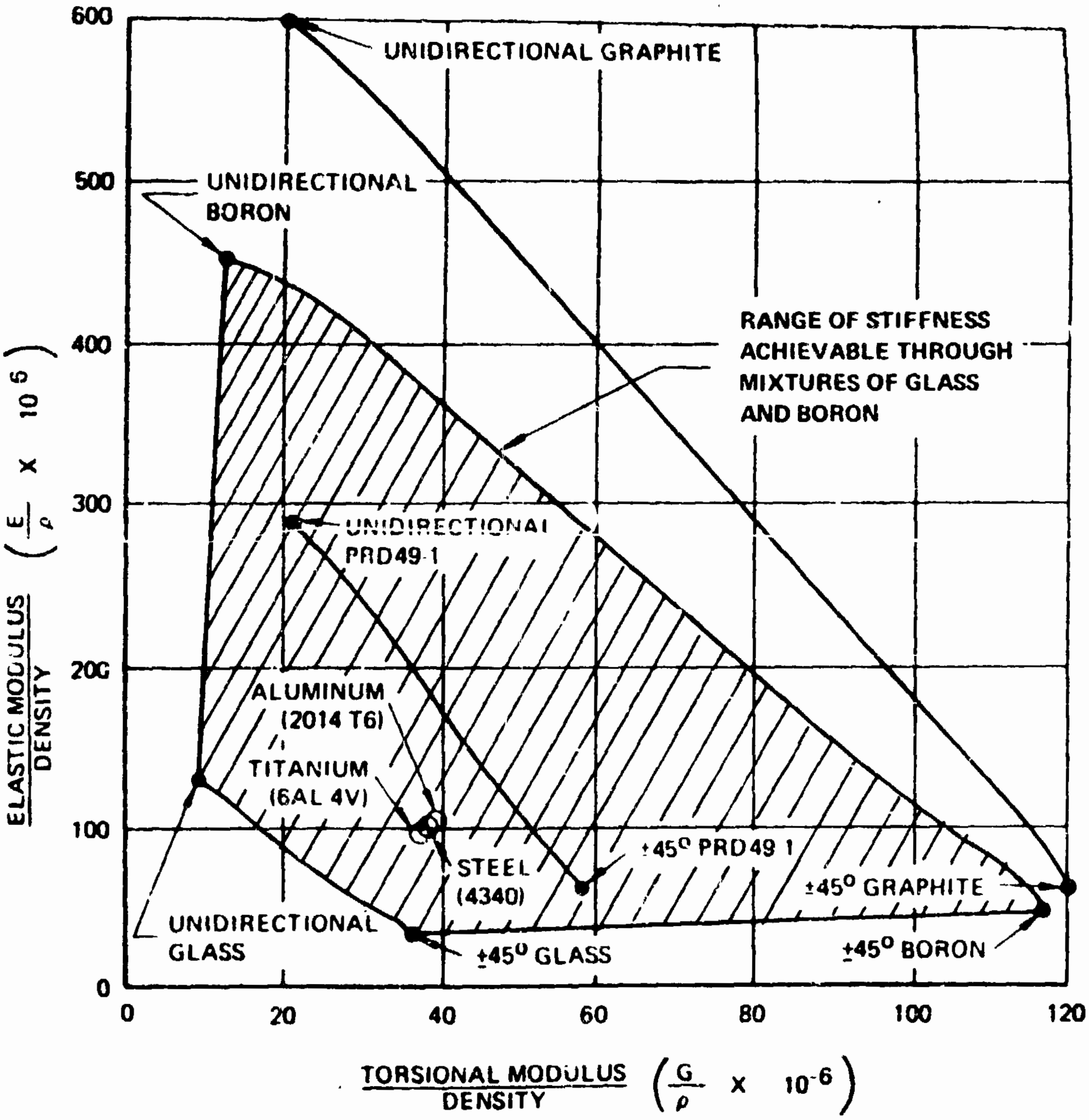



Figure 4-6. Composite Materials Provide Design Flexibility

fatigue strength to ultimate strength of advanced composites exhibited by boron and graphite offers a major advantage in increasing the fatigue strength of the aircraft. Not only is weight saved, but reduced maintenance costs are anticipated due to a significant reduction of in-service fatigue problems. The advantage of composite materials over aluminum for fatigue is shown in Figure 4.7. A display of relative weights of fatigue-critical structures is presented in Figure 4.8. For a given design limit load factor, it is expected that most of the primary wing structure can be designed for limit and ultimate conditions if advanced composites are used, while still providing a fatigue life in excess of that used for the design of corresponding metal structures.

Damage tolerance is an important consideration along with specific properties of structural materials. The superior fracture toughness of composite materials relative to aluminum alloys is clearly indicated in Figure 4.9. It should be noted however that exposed graphite/epoxy surfaces are extremely vulnerable to impact damage under normal service conditions.

The boron/epoxy has impressive compressive qualities for use in combination with other appropriate filament-matrix materials in primarily compression loaded elements.

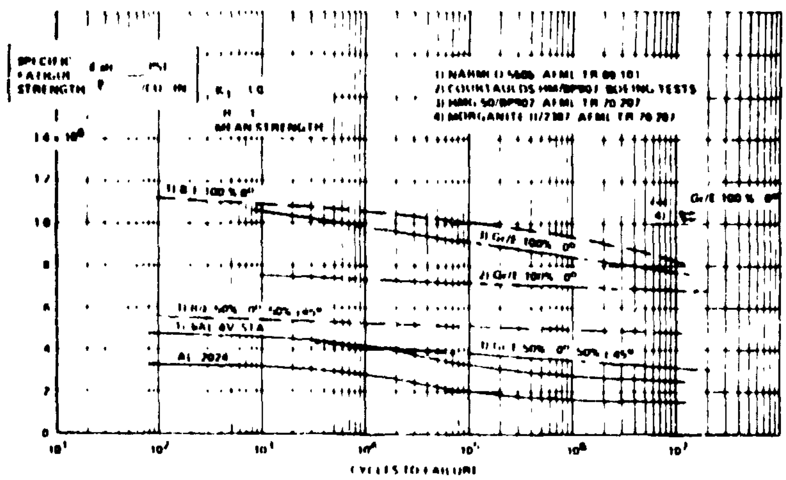


Figure 4-7. Specific Fatigue Properties

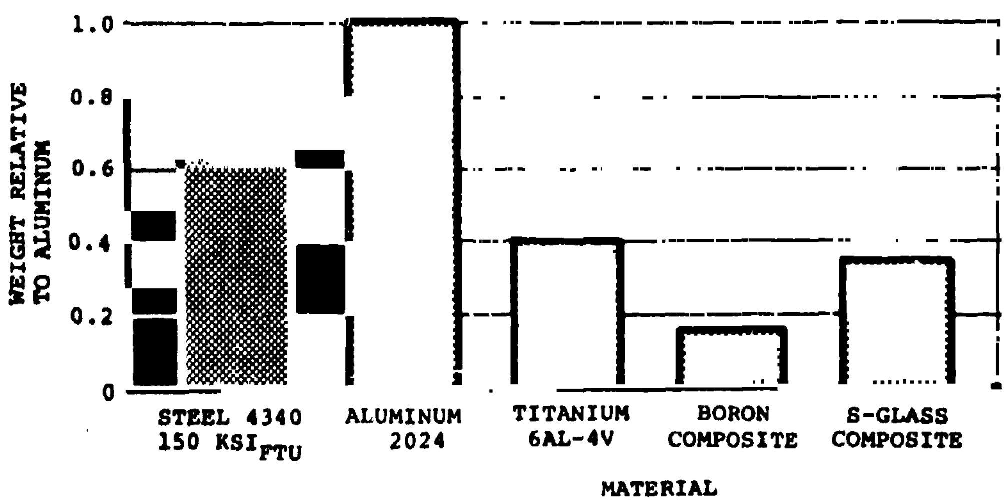



Figure 4-8. Relative Weights of Fatigue-Critical Structures

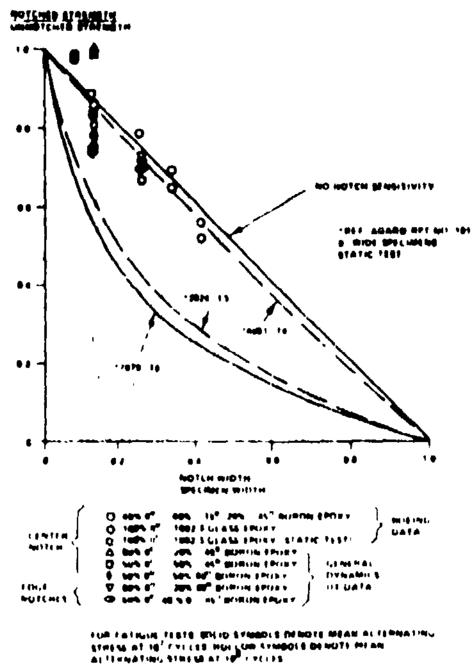


Figure 4-9. **Residual Strength of Composites and Aluminum Alloys**

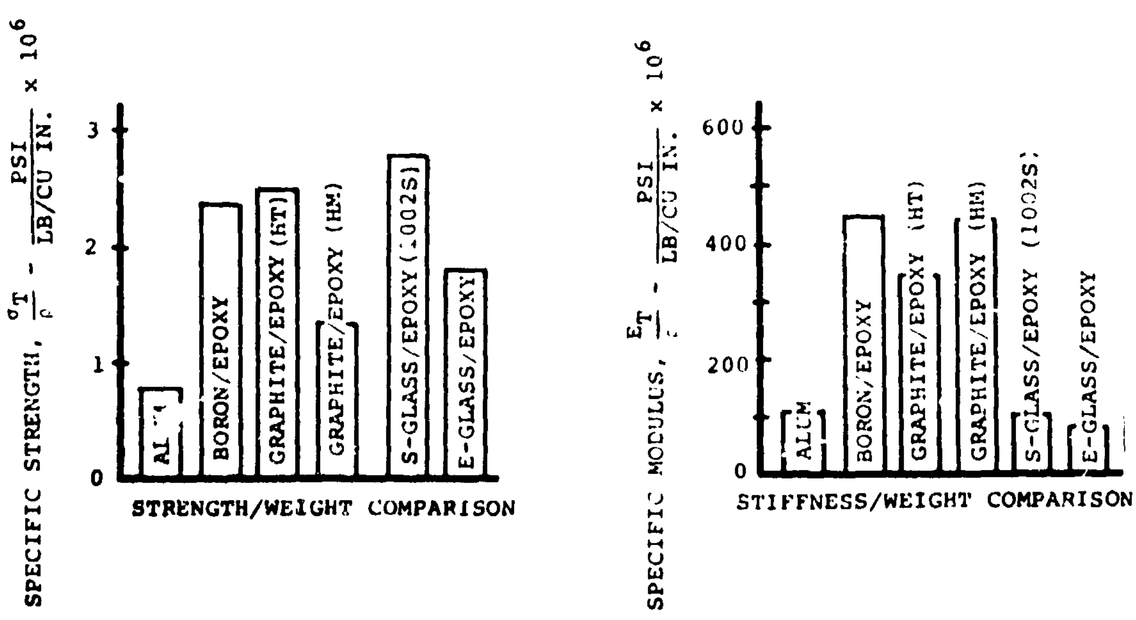



Figure 4-10. **Comparison of Material Properties**

After a careful review of the available data, boron/epoxy was selected as the basic filament matrix for use in the wing structure.

4.3.2 CORE MATERIAL

For sandwich panels, aluminum, glass, and Nomex (a nylon variant) were examined as possible core materials. Aluminum core, is extremely vulnerable to major damage by lightning when combined with either graphite or boron face sheets. This situation can readily be alleviated with current design approaches, but effort was focused on replacing aluminum as a prime candidate for the core material. Nomex is the primary candidate from an environmental resistance consideration; however, its phenolic binding is extremely vulnerable to fuel exposure. Hence for this study, a glass core has been selected.

## 4.4 WING DESIGN CONSIDERATIONS

### 4.4.1 ENGINEERING APPROACH

Current manufacturing capabilities and processes in the field of advanced composites give the designer considerable latitude in arriving at an appropriate structural configuration for the basic load-carrying element of the wing, the torque box.

In general, the advantages translate into fewer parts, a reduced number of mechanical fasteners, and an associated weight reduction, and thus, reduced manufacturing manhours.

The wing provides support for the rotor/transmission/engine combination at its extreme ends. The total fuel capacity of the tilt rotor aircraft is carried in the wings outboard of the fuselage. The propulsion units at the extreme ends of the wing are connected by a cross shaft running through the upper center portion of the torque box.

In addition to the basic problem of configuring the wing box section, the tilt rotor has joint design requirements which encompass both fixed-wing and rotary-wing technology. The design effort was focused on three main areas in the wing:

- o Basic structural shell (torque box)
- o Wing-fuselage joint
- o Joints (hardpoints)

Emphasis was directed toward:

- o Reducing the number of parts and tools
- o Reduction in machining operations

These are achieved respectively by:

- A discrete application of composite filaments/laminates
- Use of adhesives
- Pressure-molding techniques

### 4.4.2 BASIC STRUCTURAL SHELL (TORQUE BOX)

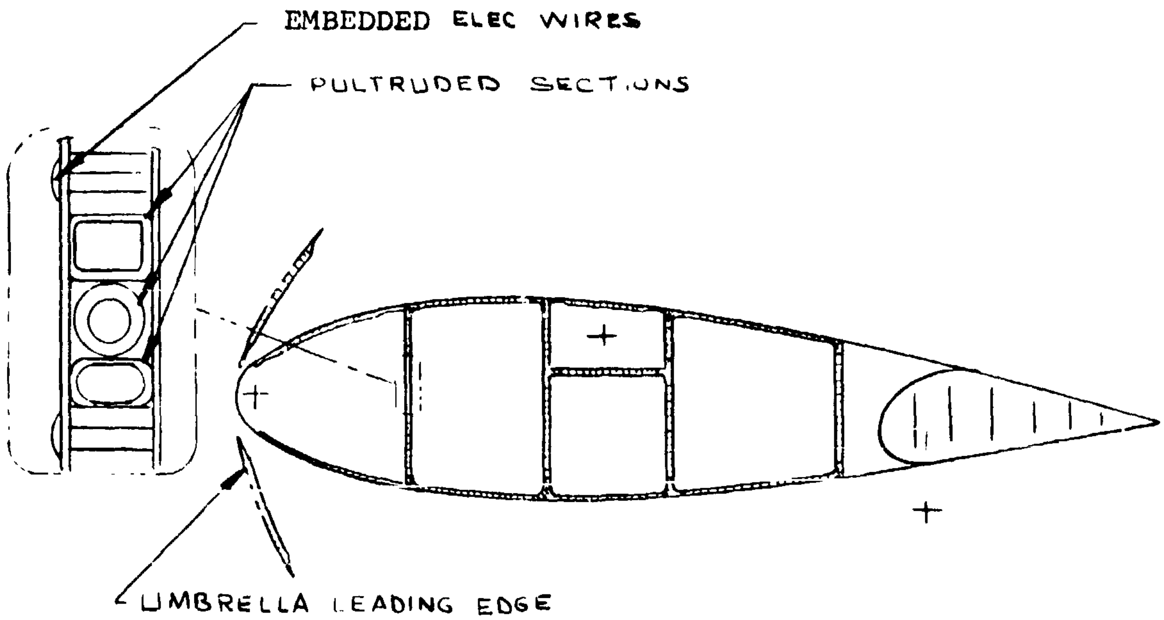
Two concepts for the basic wing torque box were considered (see Figure 4.11). One is a four-spar configuration and the other a two-spar configuration; these will be referred to as concepts A and B respectively.

Concept A - The four-spar configuration is oriented toward minimizing the number of heat cycles during the manufacturing process. The primary aim is to achieve a co-cured assembly; i.e., a one-cycle heat exposure operation. The inclusion of ribs, however, prevents this goal from being attained.

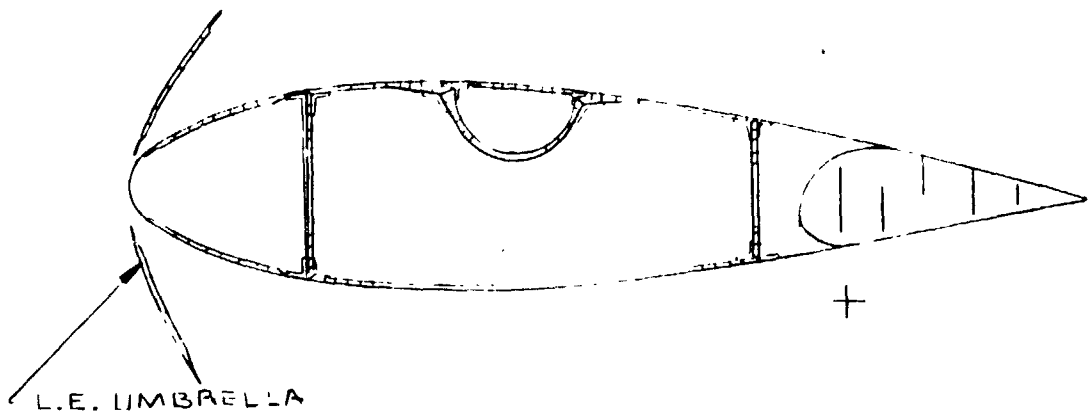
Since the inclusion of ribs presents not only manufacturing difficulties but also design problems, ribs will be provided only at flap hinges and leading-edge umbrella hinges. Intermediate ribs to react panel crushing loads will be eliminated.

The decision to eliminate intermediate ribs was based on:

- Relatively short span of the wing
- Required panel compression strength obtainable for relatively minor weight penalties




CONCEPT A




CONCEPT B

**Figure 4-11.** Basic Wing Torque Box

- No external loads being applied at the intermediate rib locations

Although the reduction in the number of curing cycles so as to approach co-curing in the manufacturing process is highly desirable, the concept A configuration has design considerations for which extensive development is required:

Too Many Access Holes

- Access required for inspection at three locations chordwise (one in each spar bay) at about 36 inches on center for length of span. Each hole has to be at least 5 inches in diameter. Providing for lightening holes in the lower portion (below cross shaft channel) of the center two spars or using a trussed configuration could eliminate access holes in the forward and aft bays, requiring them only in the center bay.
- Access hole at inboard tank end rib has to be big enough to allow for the installation of a fuel boost pump.
- Access cutouts limit area for locating chordwise filaments, if these are required. The wing is primarily loaded in spanwise bending, spanwise shear, and torsion; chordwise loads are negligible.

Ribs are difficult to install, especially the tank end rib at the inboard location.

The deletion of intermediate ribs may require baffles to reduce fuel slosh.

If the center two spars do not have lightening holes or are not of a truss configuration, cutouts are needed between the forward, center, and aft bays for fuel drainage and air venting. This means interrupting spar chords and webs in the spanwise direction (high axial load in members attaching to the spar could cause peeling problems).

Concept B - The two-spar, multirib (25-inch spacing) configuration (Figure 4.11), in general, exhibits:

° Ease of assembly

° A provision for good dimensional control and tolerance washout

° Good access is provided for inspection of structure and maintenance of systems inside the wing

° All structure is used efficiently and is multipurpose. Ribs are used for fuel baffles and to carry structural loads, spars carry structural loads and serve as integral fuel tank walls, etc.

Although the method for the design and fabrication of detail parts and assembly are within the state of the art, the incorporation of this technology in a wing design has not yet been demonstrated on a flightworthy vehicle.

Further development is needed to:

- Reduce the number of heat cycles required in construction of detail parts and assembly
- Reduce the number of parts requiring hand layup. Use of pultruded components for spar-to-rib and spar-to-panel joints should be investigated.

Based on the above discussion concept B has been chosen as a conservative approach to the design of the composite wing.

### 4.4.3 WING-FUSELAGE JOINT

The wing-fuselage joint depends on the configuration of the individual components. Assembly and field replaceability requirements virtually eliminate adhesive bonding.

A mechanical fastener design is shown in Figure 4.12. This concept is currently being developed for Boeing Vertol's Heavy Lift Helicopter. Figure 4.13 is a photograph of the fitting and Figure 4.14 a general arrangement drawing. Figure 4.15 shows the HLH application. This method offers high strength capability with minimum weight and relatively simple tooling requirements. The barrel nut installation eliminates eccentricities by placing the load path directly on the centerline of the sandwich fuselage structure. Four attachment locations using four bolts, or eight for fail safety, could provide the load paths for all the wing-to-fuselage loads.

### 4.4.4 WING-NACELLE INTERFACE

A design concept of the wing tip fitting structure is shown in Figure 4.16. This design envisions compression molding of a basic chopped-fiber element (truss) reinforced with unidirectional tape/fiber elements.

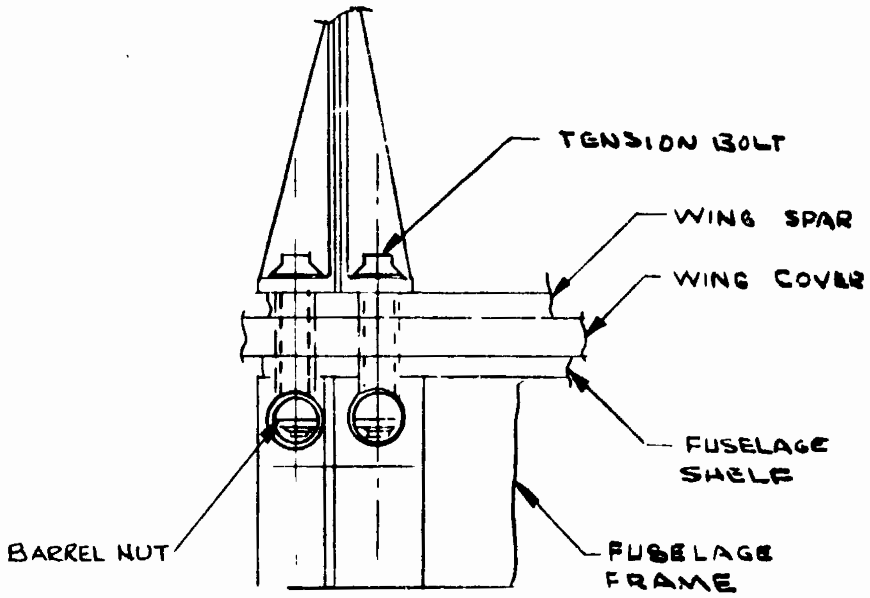



Figure 4-12. Mechanical Wing-Fuselage Joint Concept

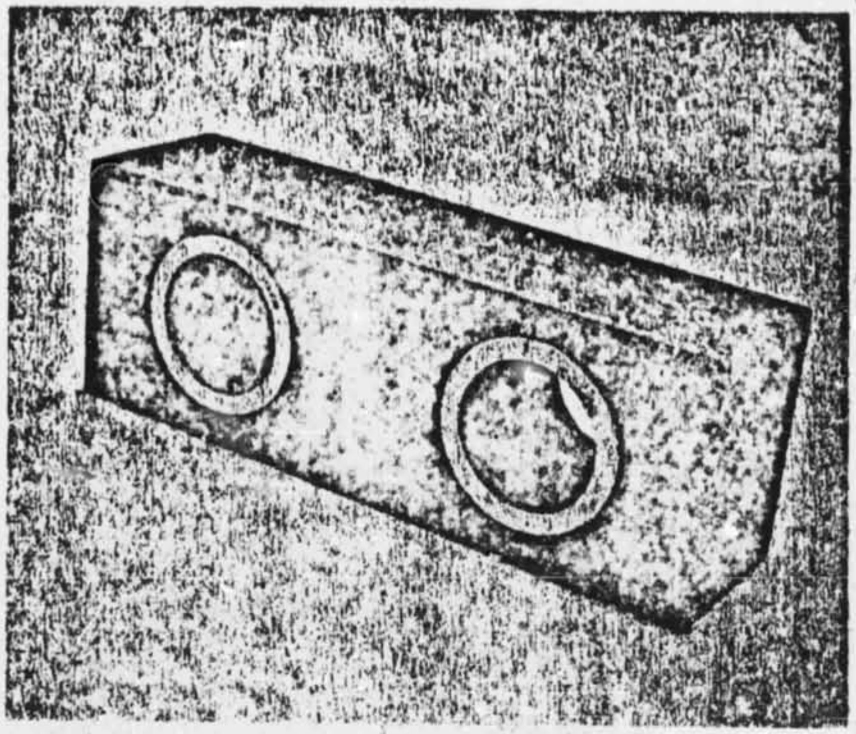


Figure 4-13. **Test Specimen for HLH Transmission Support Fitting**

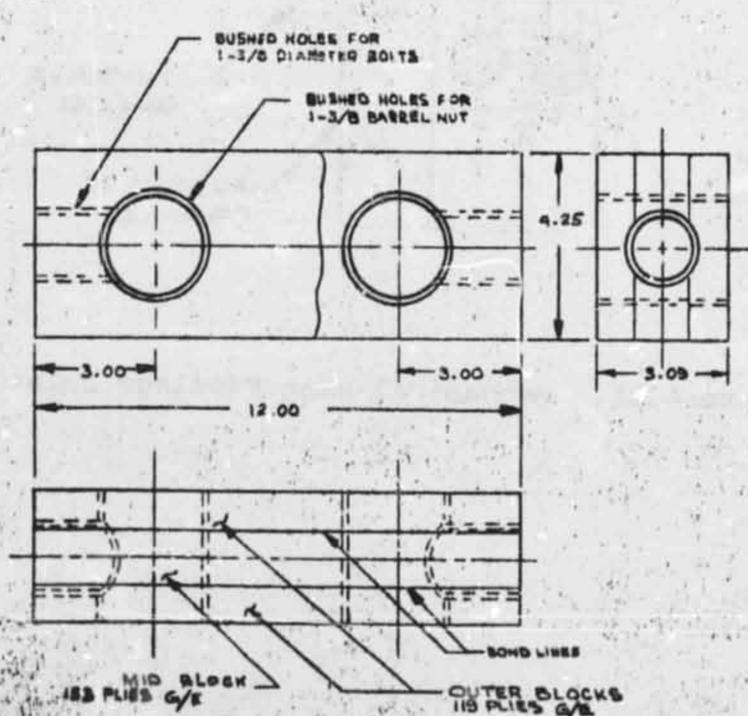



Figure 4-14. **Design of Test Specimen for HLH Transmission Support Fitting**

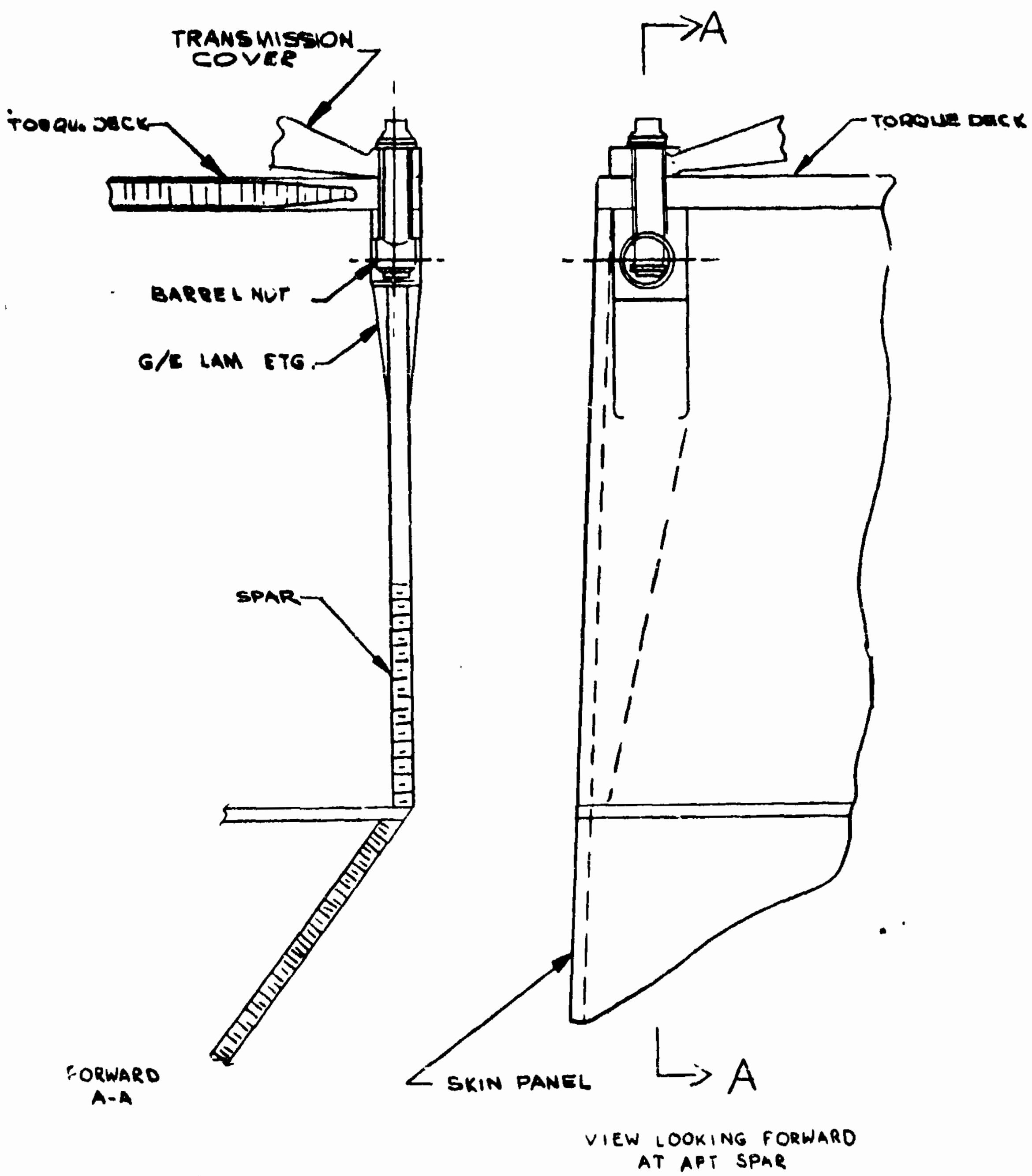



Figure 4-15. Transmission Support Fitting Concept for HLH Application

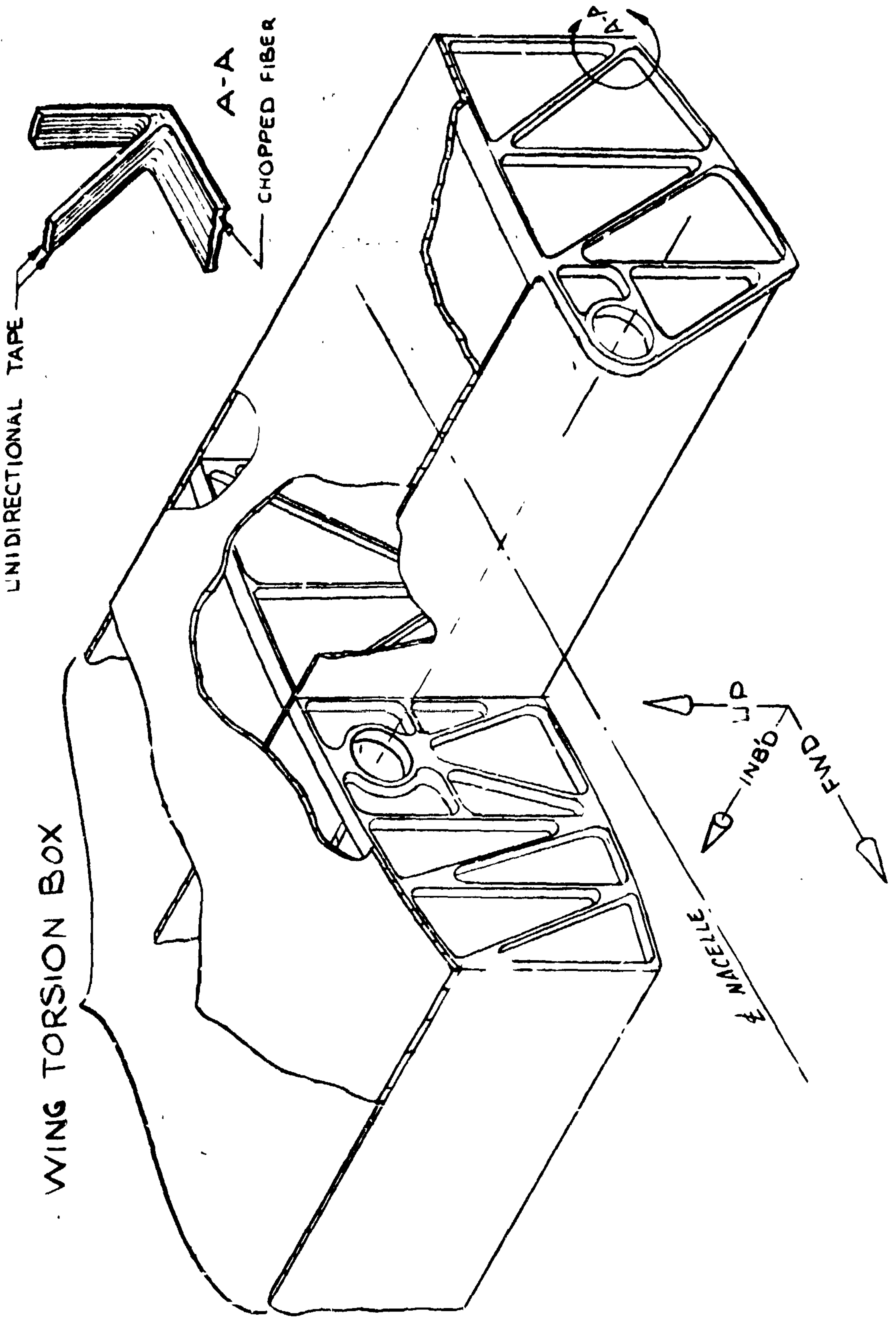



Figure 4-16. Wing-Nacelle Interface

### 4.4.5 HARDPOINTS

#### 4.4.5.1 Tension

In the event that a multibolt attachment of wing to the fuselage is employed (more efficient load transfer in a honeycomb structure), shear and tension bolt joints need to be investigated.

Figure 4.17 depicts a tension bolt concept which is capable of transmitting wing bending loads (converted to axial loads-lb./in.) for both concentrated and uniform load paths. The joint configuration shown is now being evaluated for the HLH for a major field splice. These fittings will be exposed to a fatigue environment in addition to establishing an ultimate load transfer capability.

#### 4.4.5.2 Shear

As previously stated, metals are fatigue-critical while advanced composites (graphite and boron/epoxy) are not. The quantitative extent of this advantage has yet to be established and demonstrated. A current Boeing-Vertol program has yielded preliminary results for a shear joint concept which indicate that the advantage is considerable when comparing $\sigma_e/\rho$ parameters of steel and graphite.

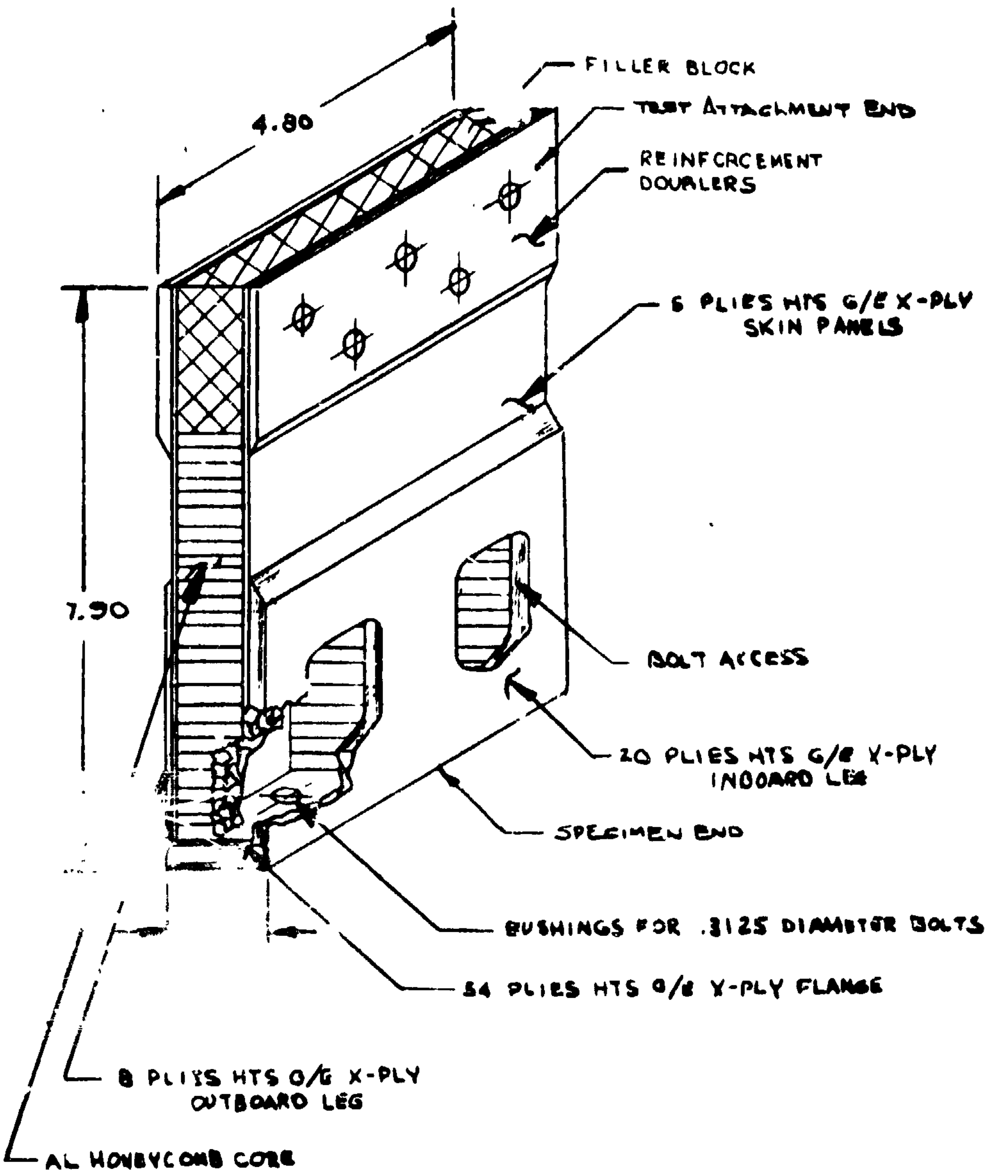



Figure 4-17. Construction of Final Tension Joint for HLH With Static Tensile Strength of 14,400 Pounds

A preliminary test specimen of 0.4-inch thickness fabricated from Hercules 2002T Gr/E (HTS/BP901) and subjected to a fatigue environment has outperformed its mating steel clevis of equal width and a total thickness of 0.7 inch. The steel clevis failed at the net tension section. When comparing $tx_\rho$ parameters, graphite/epoxy laminate has a weight advantage over the steel in the order of 10.

At present the number of tests is statistically insufficient for determination of joint design allowables. However, the tests do indicate the magnitude of the impact that the use of advanced composites will have in fatigue-critical structures.

## 4.5 STRESS ANALYSIS

### 4.5.1 SECTION PROPERTIES

Assume uniform thickness of material for the shell, fully effective in bending. See Figure 4.18 for geometry.

| Element | $\ell$ In. | $y$ In. | $\ell y$ In.$^2$ | $\ell y^2$ In.$^3$ | $I_o/t$ In.$^3$ | $I'/t$ In.$^3$ |
|---|---|---|---|---|---|---|
| 1 | 12.5 | 8 | 100 | 800 | — | |
| 2 | 4.87 | 5.93 | 28.88 | 171.4 | 9.66 | |
| 3 | 7.5 | 3.5 | 26.25 | 92 | — | |
| 4 | 4.75 | 5.88 | 27.9 | 164 | 8.92 | |
| 5 | 22.25 | 6.5 | 145.0 | 942 | — | |
| 6 | 4.16 | 2.08 | 8.66 | 18 | 6 | |
| 7 | 2.78 | -1.39 | -3.87 | 5.4 | 1.79 | |
| 8 | 22.25 | -4.5 | -100 | 450 | — | |
| 9 | 7.5 | -5.62 | -42.2 | 233 | — | |
| 10 | 12.5 | -5.58 | -69.7 | 389 | — | |
| 11 | 4.86 | -2.43 | -11.8 | 28.7 | 9.6 | |
| 12 | 6.95 | 3.475 | 24.1 | 83.7 | 28.93 | |
| $\Sigma$ | 112.87 | 1.181 | 133.22 | 3382.2 | 64.90 | 3447.1 |

$$\therefore \quad I/t = 3447.1 - 112.87\,(1.181)^2$$

$$= 3447.1 - 144.1 = 3303.0 \text{ In.}^3$$

Area for Torsion = 471 In.$^2$

$$\therefore \quad J/t = \frac{4A^2}{\Sigma \ell} = \frac{4(471)^2}{112.87} = 7860 \text{ In.}^3$$

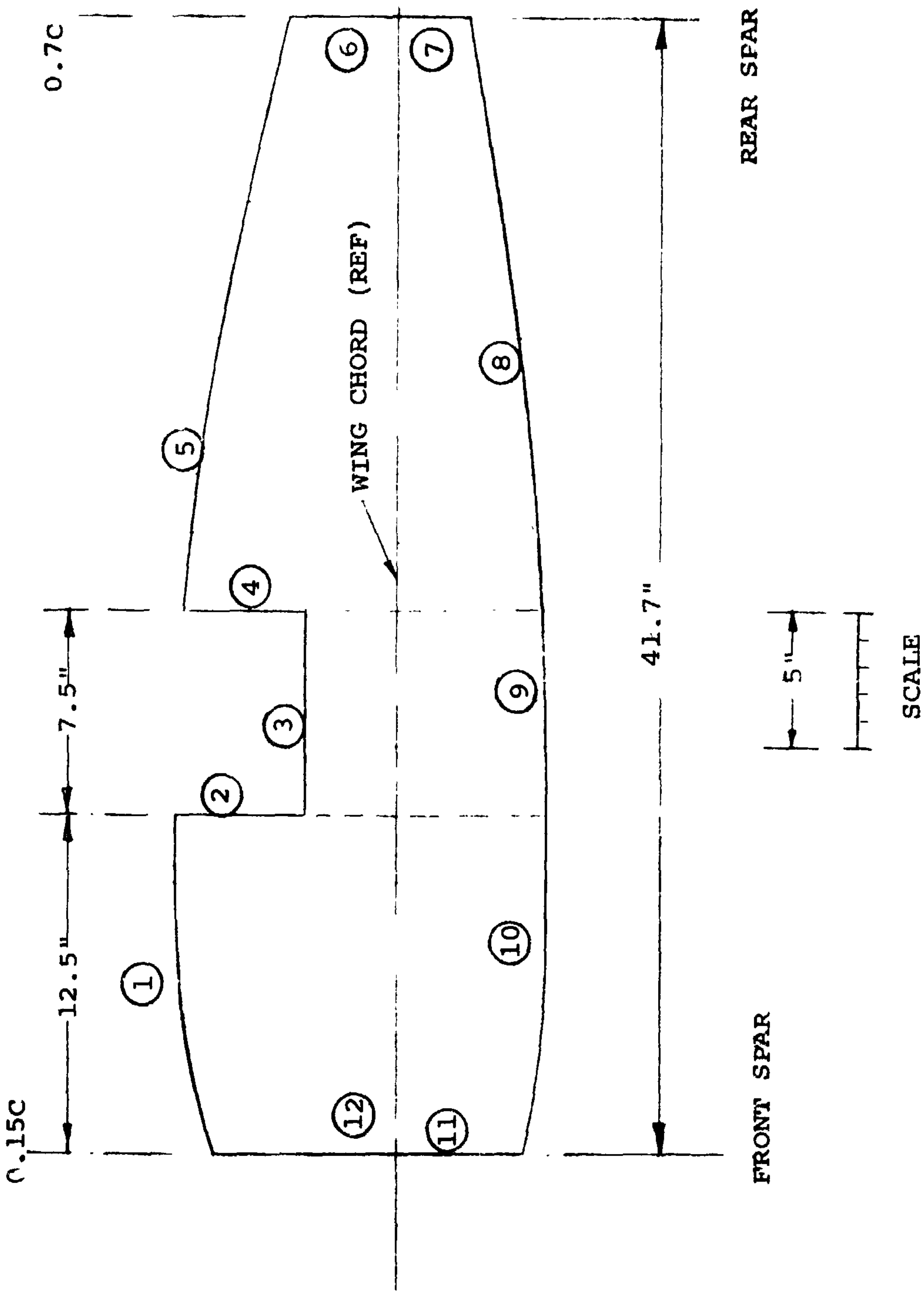



FIGURE 4.18. BASIC GEOMETRY - WING BOX

4.5.1.1 Preliminary Check for Stiffness

Stiffness Required at Root:

$$EI = 7960 \times 10^6 \text{ Lb In}^2$$

$$GJ = 1600 \times 10^6 \text{ Lb In}^2$$

Using the E and G values for boron assuming $x$ = fraction of unidirectional boron

$$10^6 \left\{30 x + 3 (1 - x)\right\} 3303t = 7960 \times 10^6$$

$$\text{i.e.} \quad (27 x + 3)t = 2.41 \qquad \ldots\ldots \text{(1)}$$

$$\text{and } 10^6 \left\{8.8 (1 - x) + 1 (x)\right\} 7860t = 1600 \times 10^6$$

$$\text{i.e.} \quad (8.8 - 7.8 x)t = .204 \qquad \ldots\ldots \text{(2)}$$

Solving above equations yields

$x$ = .846

$t$ = .0935"

$t_o$ = .079"

$t_{\pm 45}$ = .0145

As one layer of boron is .007" thick and the cross ply has to go in pairs, assume 11 plies of uni and 4 plies at $\pm 45°$ for practical design

then $x = .734$; $t = .105"$

$$EI = \left\{(27 \times .734) + 3\right\} (3303) (.105) \times 10^6$$

$$= 7900 \times 10^6 \text{ Lb. In.}^2$$

$$GJ = (8.8 - 7.8 \times .734) (7860 \times .105) \times 10^6$$

$$= 2540 \times 10^6 \text{ Lb. In.}^2$$

Stiffness Required at Tip Sta. 180:

$EI = 6450 \times 10^6$ Lb. In.$^2$

$GJ = 1600 \times 10^6$ Lb. In.$^2$

Proceeding as before

$(27\chi + 3)t = 1.95$

$(8.8 - 7.8\chi)t = .204$

$\therefore \quad \chi = .8$

$t = .0793$

Assume $t_o = .063$ 9 plies at 0°

$t_{\pm 45} = .028$ 4 plies at $\pm 45$

$t = .091$

$\chi = .692$

$\therefore \quad EI = \left\{(27 \times .692) + 3\right\}(3303)\ (.091) \times 10^6 = 6520 \times 10^6$ Lb. In.$^2$

$GJ = \left\{8.8 - (7.8 \times .692)\right\}(7860)\ (.091) \times 10^6 =$

$2430 \times 10^6$ Lb. In.$^2$

## 4.5.2 PANEL INSTABILITY

Compression Panels

The critical axial loading for compression panels with different lay-ups are computed below and shown in Figure 4.19:

Layup A $\quad n$ at $0 = 12 = .667$

$n$ at $\pm 45 = 4 = .222$

$n$ at $90 = 2 = .111$

$\therefore \quad t_f = 0.5 \times 18 \times .007 = .063"$

$c \;= \text{core thickness}$

$$10^{-6}E_X = (30 \times .666 + 3 \times .222 =)\ 20.67\ \text{Lb./In.}^2$$

$$10^{-6}E_Y = (30 \times .111 + 3 \times .222 =)\ 4.00\ \text{Lb./In.}^2$$

$$10^{-6}G = (.666 + .111 + 8.8 \times .222 =)\ 2.73\ \text{Lb./In.}^2$$

Assume $\mu_{XY} = .4$

$$\mu_{YX} = .4 \times \frac{4.0}{2067} = .078$$

$$\therefore \quad \psi = \frac{E}{\lambda} = \sqrt{\frac{20.67 \times 4.00}{1 - .4 \times .078}} \times 10^6$$

$$= 9.25 \times 10^6\ \text{Lb./In.}^2$$

The allowable load/in. is given by

$$N_{XCR} = \frac{k_x \ \pi^2 \ \psi \ t_f \ c(c + t_f)}{2b^2}$$

For a rib spacing of 25" $\;K_x = 2.5$

$$\therefore \quad N_{XCR} = \frac{2.5\pi^2 \times 9.25 \times 10^6 \times .063c(c + .063)}{2b^2}$$

$$= 7.2 \times 10^6 \ \frac{c(c + .063)}{b^2}$$

Evaluating as function of b and c

| b (inch) | | 10 | 20 | 30 | 40 |
|---|---|---|---|---|---|
| | | | $N_{XCR}$ Lb./In. | | |
| c (inch) | = .3 | 7850 | 1960 | 872 | 491 |
| | .5 | 20300 | 5060 | 2250 | 1266 |
| | .6 | 28600 | 7160 | 3180 | 1790 |
| | .7 | 38450 | 9620 | 4270 | 2400 |

$N_{XU}$ (Ult) $= .126 \times (300 \times .666 + 30 \times .222 + 25 \times .111) \times 10^3$

$= 26400$ Lbs./In.

Layup B (Layup A + 2 Laps at 90°)

$\therefore \; n_o = 12 = .6$

$n_{\pm 45} = 4 = .2$

$n_{90} = 4 = .2$

$t_f = 0.5 \times 20 \times .007 = .07"$

$10^{-6} E_X = (30 \times .6 + 3 \times .2 =)\ 18.6$ Lb./In.$^2$

$10^{-6} E_Y = (30 \times .2 + 3 \times .2 =)\ 6.6$ Lb./In.$^2$

$10^{-6} G = (.6 + .2 + 8.8 \times .2 =)\ 2.56$ Lb./In.$^2$

$\mu_{XY} \quad 0.42$

$\mu_{YX} = 0.42 \times \frac{6.6}{18.6} = 0.149$

$\therefore \quad = 10^6 \times \sqrt{\frac{18.6 \times 6.6}{1 - (.42 \times .149)}} = 11.42 \times 10^6$ Lb./In.$^2$

$$N_{XCR} = \frac{2.5\,\pi^2 \times 11.42 \times 10^6 \times .07c(c + .07)}{2b^2}$$

$$= 9.87 \times 10^6 \times \frac{c(c + .07)}{b^2}$$

Evaluating $N_{XCR}$

| b (inch) | | 10 | 20 | 30 | 40 |
|---|---|---|---|---|---|
| | | | $N_{XCR}$ | Lb./In. | |
| c (inch) | = .3 | 10980 | 2740 | 1220 | 685 |
| | .5 | 28200 | 7040 | 3130 | 1760 |
| | .6 | 39700 | 9920 | 4410 | 2480 |
| | .7 | 53200 | 13300 | 5920 | 3320 |

$N_{XU}$ (Ult) $= .14\ (300 \times .6 + 30 \times .2 + 25 \times 2) \times 10$

$= 26750$ Lbs./In.

Layup C (Layup A + 4 Laps at 90°)

i.e., $n_o = 12 = .545$

$n_{\pm 45°} = 4 = .182$

$n_{90} = 6 = .273$

$t_f = 0.5 \times 22 \times .007 = .077"$

$10^{-6}\ E_X = (30 \times .545 + 3 \times .182 =)\ 16.89$ Lb./In.$^2$

$10^{-6}\ E_Y = (30 \times .273 + 3 \times .182 =)\ 8.74$ Lb./In.$^2$

$10^{-6}\ G = (545 \times .273 + 8.8 \times .182 =)\ 2.418$ Lb./In.$^2$

$\mu_{XY} = 0.42$ (Assumed)

$$\mu_{YX} = 0.42 \times \frac{8.74}{16.89} = 0.218$$

$$\therefore \quad = 10^6 \times \sqrt{\frac{16.89 \times 8.74}{1 - .42 \times .218}} = 12.78 \times 10^6 \text{ Lb./In.}^2$$

$$N_{XCR} = \frac{2.5\,\pi^2 \times 12.78 \times 10^6 \times .077(c + .077)c}{2b^2}$$

$$= 12.12 \times 10^6\ \frac{c(c + .077)}{b^2}$$

Evaluating $N_{XCR}$ as before

| b (inch) | | 10 | 20 | 30 | 40 |
|---|---|---|---|---|---|
| | | | $N_{XCR}$ | Lb./In. | |
| c (inch) | = .3 | 13720 | 3430 | 1525 | 858 |
| | .5 | 35000 | 8750 | 3885 | 2188 |
| | .6 | 49400 | 12350 | 5490 | 3080 |
| | .7 | 66000 | 16500 | 7340 | 4125 |

$N_{XU}$ (Ult) $.154(300 \times .545 + 30 \times .182 + 25 \times .273) \times 10^3$

$= 27050$ Lbs./In.

Shear Buckling

Assume V = 0.1

Then $F_S = \frac{\pi^2 K}{4} \left(\frac{h}{b}\right)^2 \frac{E'}{\lambda}$

Where $E = \sqrt{E_X E_Y}$

$\lambda = \sqrt{1 - \mu_{XY} \mu_{YX}}$

$h = c + t_f$

$a = 25"$

Layup A

$\frac{E'}{\lambda} = 9.25 \times 10^6$ Lb./In.$^2$

$F_{SCR} = 22.8 \times 10^6 \, K \left(\frac{h}{b}\right)^2$

$t_f = .063$

$q_{CR} = 2.87 \times 10^6 \, K \left(\frac{h}{b}\right)^2$

For b > a use value of a for b in equation

| | | | | | |
|---|---|---|---|---|---|
| b (inch) | | 10 | 20 | 30 (25) | 40 (25) |
| b/a | | .4 | .8 | 1.2 (.833) | 1.6 (.625) |
| $K_M = K$ | | 2.9 | 3.5 | 5.25 | 4.8 |
| | | $q_{CRIT.}$ Lb./In. | | | |
| c = (inch) .3; | h = (inch) .363 | 10950 | 3320 | 3190 | 2900 |
| .5; | .563 | 26400 | 8000 | 7660 | 6980 |
| .6; | .663 | 36500 | 11000 | 10620 | 9690 |
| .7; | .763 | 48500 | 14700 | 14100 | 12850 |

$$q_{ULT} \quad .126 \times 10^3 \ (9 \times .666 + 9 \times .111 + 67 \times .222)$$

$$= 2760 \text{ Lb./In.}$$

Layup B

$$\frac{E'}{\lambda} = 11.42 \times 10^6 \text{ Lb./In.}^2$$

$$F_{SCR} = 28.2 \times 10^6 \ K \left(\frac{h}{b}\right)^2$$

$$t_f = .07$$

$$q_{CR} = 3.95 \times 10^6 \ K \left(\frac{h}{b}\right)^2$$

K Values As For Layup A

$$_{ULT} = .14 \times 10^3 \ (9X.6+9X.2+67X.2) = 2880 \text{ Lb./In.}$$

Layup C

$$\frac{E'}{\lambda} = 12.78 \times 10^6 \text{ Lb./In.}^2$$

$$t_f = .077$$

$$q_{CR} = 4.86 \times 10^6 \ K \left(\frac{h}{b}\right)^2$$

K Values As For Layup A

$$q_{ULT} = .154 \ (9X.545+9X.273+67X.182)X10^3 = 3010 \text{ Lb./In.}$$

$$q_{CR} > q_{ULT} \text{ for } c > 0.3 \text{ and } b \leq 40''$$

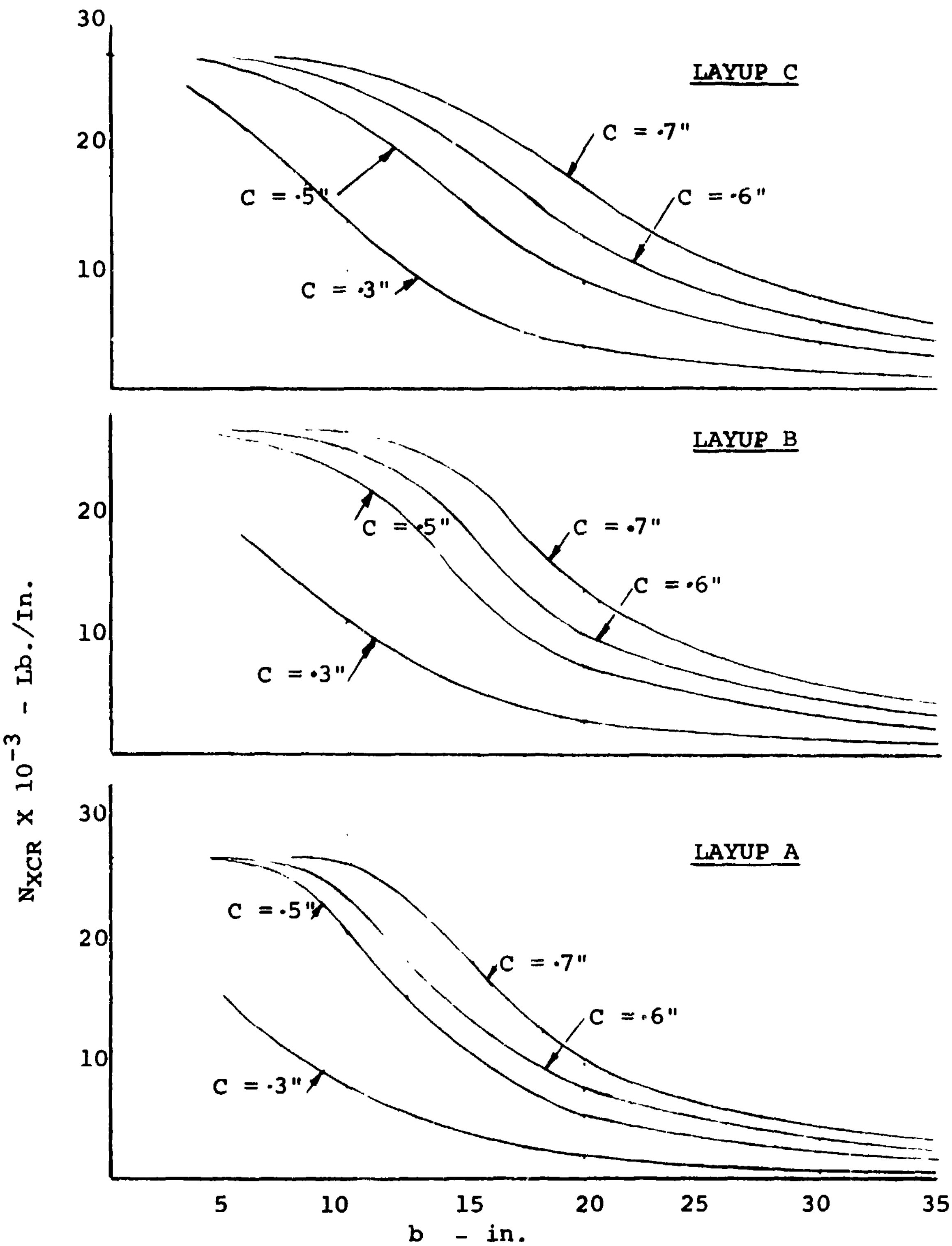



Figure 4.19. Allowable Load/In. - Compression Panels
(B/E Honeycomb Sandwich)

### 4.5.3 STRENGTH CHECK

Root (Wing Sta. 30)

1. Axial Loads

$BM = 4.32 \times 10^6$ In. Lb.

$I/t = 3303$ In.$^3$

| | | Distance From Front Spar (Inch) | | | | | |
|---|---|---|---|---|---|---|---|
| | | (F.S.) 0 | 10 | 15 | 20 | 30 | (R.S.) 41.7 |
| $Y_U$ | In. | 5.769 | 7.279 | 2.319 | 7.019 | 5.619 | 2.979 |
| $Y_L$ | In. | 6.041 | 6.801 | 6.801 | 6.801 | 5.841 | 3.961 |
| $N_{XU}$ | Lb./In. | -7560 | -9540 | -3040 | -9200 | -7360 | -3900 |
| $N_{XL}$ | Lb./In. | 7910 | 8910 | 8910 | 8910 | 7650 | 5190 |
| F (Shear Lag Factor) | | 1.35 | .97 | .87 | .9 | 1.02 | 1.35 |
| $N_{XU}$ | Lb./In. | -10200 | -9240 | -2650 | -8280 | -7410 | -5860 |
| $N_{XL}$ | Lb./In. | 10690 | 8640 | 7840 | 8020 | 7800 | 7000 |

2. Shear Loads

$V = 16200$ Lb.

$T = \pm 200{,}000$ In. Lb. (Assumed)

$2A = 942$ In.$^2$

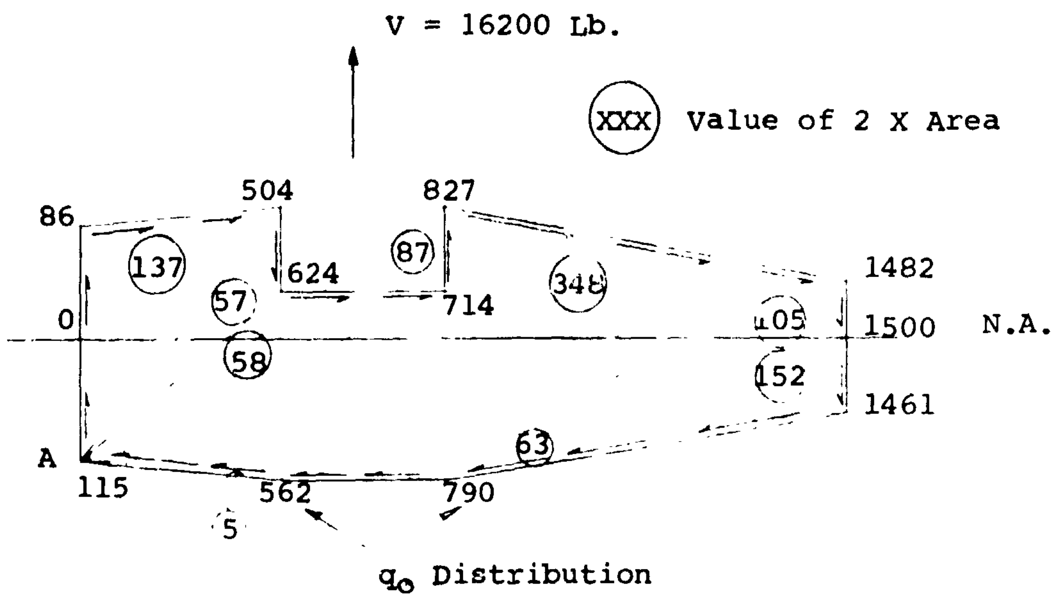
Basic Shear Flow $q = q_o + q_1 + q_T$ Lb./In.
(Without Shear Lag)

where $q_o$ = shear flow in cut structure

$q_1$ = balancing shear flow

$q_T$ = shear flow due to torque

Basic Shear Distribution




$\Sigma F_Z$ = 250+454-2760+3620-4740-4450-5860-3230+253+347+16200

= 21124 -21040 = (84)

Taking Moments About $P_T$ 'A'

M = 295X137+564X57+669X58 - 771X87+1155X348+1491X105

+1480X152+1126+676X5

= 40400+32150+38800-67200+402000+156900+223500+71000+3480

= 901030 In. Lb.

$2A_{EFF}$ = 137+57+58+87+348+105+152+63+5-70

= 942 In.$^2$

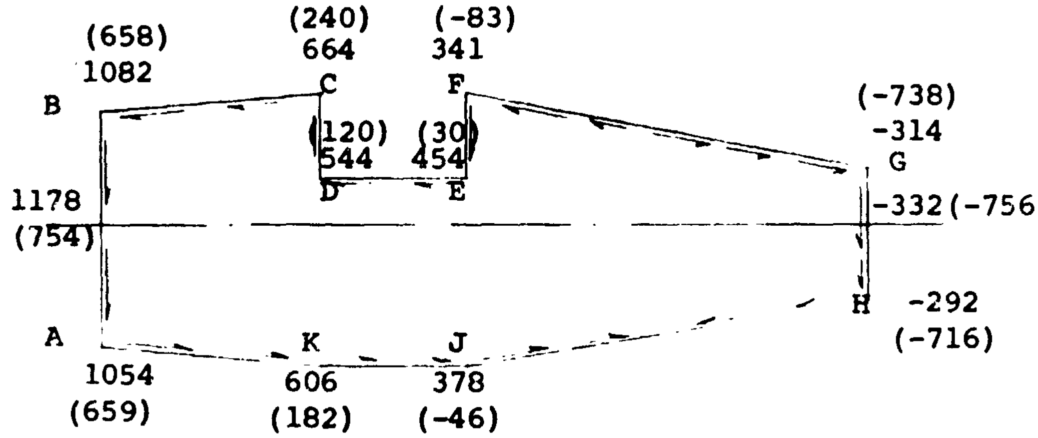
$$q_1 = \frac{901030}{942} = 956 \text{ Lb./In.}$$

$$q_T = \frac{\pm 200000}{942} = \pm 212 \text{ Lb./In.}$$

q (without shear lag effects)

$= q_o - q_1 \pm q_T$

Values for R.H. Wing Shown in Parenthesis




Shear Lag Effects

(Average Shear Flows)

| Member | $q_V$ (Due to V No Shear Lag) | Shear Lag Factor G | $q_V = Gq_V$ | $q_T$ | $q_{LH}$ | $q_{RH}$ |
|---|---|---|---|---|---|---|
| AB | 856 | 2.75 | 2360 | ±2. | 2572 | 2148 |
| BC | 661 | 1.75 | 1160 | | 1372 | 948 |
| CD | 392 | 1.43 | 560 | | 772 | 348 |
| DE | 287 | 1.25 | 359 | | 571 | 147 |
| EF | 185 | 1.15 | 213 | | 425 | 0 |
| FG | -198 | 1.5 | -297 | | -87 | -509 |
| GH | -515 | 2.75 | -1420 | | -1208 | -1632 |
| HJ | -170 | 1.5 | -255 | | -43 | -467 |
| JK | 280 | 1.25 | 350 | | 562 | 138 |
| KA | 618 | 1.75 | 1081 | ±212 | 1293 | 869 |

Tip (Sta. 180)

BM = 1.05 X $10^6$ In. Lb.

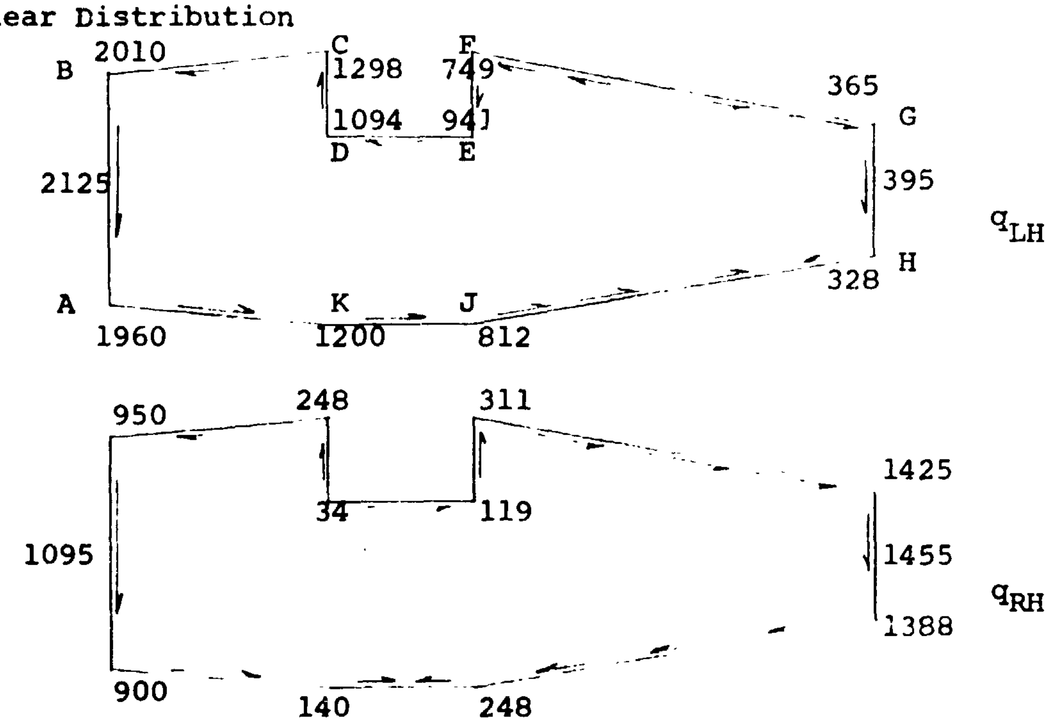
Shear = 27,500 Lb.

Torque = $\pm$500,000 In. Lb. (Assumed)

1. Axial Loads (Geometry and I/t as at root)

| | Distance From Front Spar (Inch) | | | | | |
|---|---|---|---|---|---|---|
| | (F.S.) 0 | 10 | 15 | 20 | 30 | (R.S.) 41.7 |
| $N_{XU}$ Lb./In. | -1835 | -2315 | -740 | -2230 | -1790 | -946 |
| $N_{XL}$ Lb./In. | 1920 | 2160 | 2160 | 2160 | 1860 | 1260 |

2. Shear Distribution




$q_{TORQUE}$ = $\pm$530 Lb./In.

<u>Panel BC</u>

At root select layup B; c = 0.5"

$$b = 12.0'' \text{ effective}$$

$$N_{XCR} = 20200 \text{ Lb./In.}$$

$$q_{CR} = 2880 \text{ Lb./In.}$$

$$N_X = -10200 \text{ Lb./In.}$$

$$q_{AV} = 1372 \text{ Lb./In.}$$

$$R_C = .505$$

$$R_S = .477; \; R_C + R_S = 982$$

$$M.S. = \frac{1}{.982} - 1 = 0.02$$

At tip select layup A; c = 0.5"

$$N_{XCR} = 16000 \text{ Lb./In.}$$

$$q_{CR} = 2760 \text{ Lb./In.}$$

$$N_X = -2440 \text{ Lb./In.}$$

$$q_{AV} = 1650 \text{ Lb./In.}$$

$$R_C = .152$$

$$R_S = .597$$

$$M.S. = \frac{1}{.152+.597} - 1 = 0.33$$

Panels CD, DE, and EF

Layup A will be satisfactory by comparison with panel BC

Panel FG

At root select layup C with c = 0.6"

$$b = 22\text{" effective}$$

$$N_{XCR} = 10{,}400 \text{ Lb./In.}$$

$$q_{CR} = 3010 \text{ Lb./In.}$$

$$N_X = -8280 \text{ Lb./In.}$$

$$q = 509 \text{ Lb./In.}$$

$$R_C = .796$$

$$R_S = .169$$

$$M.S. = \frac{1}{.965} - 1 = 0.03$$

At tip select layup A with c = 0.6"

$$N_{XCR} = 6200 \text{ Lb./In.}$$

$$q_{CR} = 2760 \text{ Lb./In.}$$

$$N_X = -2230 \text{ Lb./In.}$$

$$q = 311 + 2/3\ (1114) = 1058 \text{ Lb./In.}$$

$$R_C = .36$$

$$R_S = .383$$

$$M.S. = \frac{1}{.743} = 0.34$$

Lower Cover

To allow for taxi conditions, assume design compression load = .4 X Design Tension Load Case

Assume $b_{eff} = 40"$

Compression Case

At root $q \approx 400$ lb/in.

$$N_{X_{AV}} = [(10690 \times 5 + 8640 \times 10 + 8020 \times 10 + 7800 \times 10 + 7000 \times 6.7)/41.7] \times 0.4$$

$$= 3310 \text{ lb/in.}$$

Select layup C with c = 0.7"

$$N_{X_{CR}} = 4125 \text{ lb/in.}$$

$$q_{CR} = 3010 \text{ lb/in.}$$

$$R_C = .802$$

$$R_S = .133$$

$$\therefore \text{M.S} = \frac{1}{.935} = \underline{.07}$$

At tip; layup A with c = 0.7"

$$N_{X_{CR}} = 2400 \text{ lb/in.}$$

$$q_{CR} = 2760 \text{ lb/in.}$$

$$N_{X_{AV}} = \frac{.4}{41.7}[1920 \times 5 + 2160 \times 20 + 1860 \times 10 + 1260 \times 6.7]$$

$$= -765 \text{ lb/in.}$$

$$q_{AV} = 1200 \text{ lb/in.}$$

$$R_C = .323$$

$$R_S = .435$$

$$\therefore \text{M.S.} = \frac{1}{.758} - 1 = 0.32$$

Check Tension Case

At root $N_{X_{ALL}} = .154(178 \times .545 + 22 \times .182 + 10 \times .273) \times 10^3$

$= 16000$ lb/in.

$q_{ALL} = 3010$ lb/in.

$N_{X_{MAX}} = 10690$ lb/in.

$q = 1293$ lb/in.

Layup C is inadequate over forward portion - use 2 additional layers at 0° over forward 20" then

$N_{X_{ALL}'} = .168(17.8 \times .584 + 22 \times .166 + 10 \times .25) \times 10^3$

$= 18550$ lb/in.

$q_{ALL} = .168(9 \times .584 + 9 \times .25 + 67 \times .166) \times 10^3$

$= 3130$ lb/in.

$R_T = .576$

$R_S = .414$

$\therefore$ M.S. $= \frac{1}{.990} = .01$

At tip $N_{X_{ALL}} = .126(178 \times .666 + 22 \times .222 + 10 \times .111) \times 10^3$

$= 15530$ lb/in.

$q_{ALL} = 2760$ lb/in.

$N_{X_{MAX}} = 2160$ lb/in.

$q_{MAX} = 1960$ lb/in.

$R_T = .139$

$R_S = .71$

$\therefore$ M.S. $= \frac{1}{.849} - 1 = .18$

Front Spar

At Root

$$q_{MAX} = 2572 \text{ lb/in.}$$

As the spar is assumed to be effective in bending, design for an average axial load of ± 5000 lb/in.

$$h_{eff} = 12"$$

Layup D $\quad n @ 0 = 12 = .545$

$n @ \pm 45^{o} = 8 = .364$

$n @ 90 = 2 = .091$

$$t_f = 0.5 \times 22 \times .007 = .077$$

$$C = 0.7"$$

$$10^{-6}E_X = (30 \times .545 + 3 \times .364 =) \quad 17.41 \text{ lb/in}^2$$

$$10^{-6}E_Y = (30 \times .111 + .364 =) \quad 4.42 \text{ lb/in}^2$$

$$10^{-6}G = (.545 + .091 + 8.8 \times .364 =) \quad 3.84 \text{ lb/in}^2$$

$$\mu_{xy} = 0.42$$

$$\mu_{yx} = 0.42 \times \frac{4.42}{17.41} = .106$$

$$\therefore \psi = 10^6 \times \sqrt{\frac{17.41 \times 4.42}{1-(.42 \times .106)}} = 8.99 \times 10^6 \text{ lb/in.}^2$$

$$h = 12"$$

$$N_{X_{CR}} = \frac{2.5 \pi^2 \times 8.99 \times 10^6 \times .077 \times .7 \times .777}{2 \times 12^2}$$

$$= 32000 \text{ lb/in.}$$

$$N_{X_{ULT}} = .154(300 \times .545 + 30 \times .364 + 25 \times .091) \times 10$$

(Fully Stable) $= 27200$ lb/in.

Allowing for ultimate capability and effective modulus effects

use $N_{X_{CR}} = 25000$ lb/in.

with $V = 0.1$

$a = 25"$

$b = 12"$

$b/a = .48$

$K = K_M = 3$

$E'/\lambda = 8.99 \times 10^6$ lb/in.$^2$

$h = .777"$

$$q_{CR} = \frac{\pi^2 K}{4}\left(\frac{h}{b}\right)^2 \frac{E'}{\lambda}(2t_f)$$

$$= \frac{3\pi^2}{2} \times \left(.077 \times 8.99 \times 10^6\right) \left(\frac{.777}{12}\right)^2$$

$$= 43.1 \times 10^4 \text{ lb/in.}$$

$$q_{ULT} = .154(9 \times .545 + 9 \times .091 + 67 \times .364) \times 10^3$$

$$= 4450 \text{ lb/in.}$$

$\therefore R_C = .2$

$R_S = .58$

and $\quad$ $M.S. = \frac{1}{.78} - 1 = 0.28$

At tip

$q = 2125$ lb/in.

$N_{X_{AV}} = \pm 1000$ lb/in.

For layup A with C = 0.7

$q_{ALL} = 2760$ lb/in.

$N_{X_{ALL}} = 24000$ lb/in.

$R_C = .041$

$R_S = .77$

$\therefore M.S. = \frac{1}{.811} - 1 = 0.23$

Rear Spar

At root $q_{MAX} = 1632$ lb/in.

$N_{X_{AV}} = \pm 3500$ lb/in.

Use layup A with C = .6"

$q_{ALL} = 2760$ lb/in.

$N_{X_{ALL}} = 1900$ lb/in.

$R_C = .185$

$R_S = .592$

$\therefore$ M.S. $= \frac{1}{.787} - 1 = 0.28$

At tip

$q_{MAX} = 1455$ lb/in.

$N_{X_{AV}} = \pm 600$ lb/in.

Layup E

n @ 0 = 6 = .50

n @ $\pm$45 = 4 = .333

n @ 90 = 2 = .167

C = 0.6"

$t_f = 0.5 \times 12 \times .007 = .042$

Assume $\mu_{xy} = 0.4$

$\mu_{yx} = 0.4 \times \frac{6}{16} = .15$

$\therefore \psi = 10^6 \times \sqrt{\frac{16 \times 6}{1 - .4 \times .15}} = 10.1 \times 10^6$ lb/in.$^2$

$h = 12"$

$$N_{X_{CR}} = \frac{2.5\ \pi^2 \text{ X } 10.1 \text{ X } 10^6 \text{ X } .042 \text{ X } .6 \text{ X } .642}{2 \text{ X } 12^2}$$

$= 14000$ lb/in.

$N_{X_{ULT}} = .084(300 \text{ X } .50 + 30 \text{ X } .333 + 25 \text{ X } .167) \text{ X } 10^3 = 13800$ lb/in.

Use $N_{X_{ALL}} = 12000$ lb/in.

With $V = 0.1$

$a = 25"$

$b = 12"$

$b/a = .48$

$K = K_M = 3$

$E'/\lambda = \psi = 10.1 \text{ X } 10^6$ lb/in.$^2$

$h = .642"$

$$q_{CR} = \frac{3\pi^2}{2} \text{ X } .042 \text{ X } 10.1 \text{ X } 10^6 \text{ X } \left(\frac{.642}{12}\right)^2$$

$= 18000$ lb/in.

$q_{ULT} = .084(9 \text{ X } .50 + 9 \text{ X } .167 + 67 \text{ X } .333) \text{ X } 10^3$

$= 2380$ lb/in.

$R_C = .05$

$R_S = .61$

$\therefore$ M.S. $= \dfrac{1}{.66} - 1 = 0.5$

Corner Gussets

Gussets are required at all corners to carry torsional shear and at C, D, E and F to stabilize panel.

Maximum shear flow for design = 1300 lbs/in.

with $\tau_{all}$ (XP251-`S´ Glass) = 48000 psi X-Ply

$\therefore$ Av. `t´ required = .027"

Minimum practical thickness = 4 X .010 - .040"

(2, $\pm$ 45$^{o}$ layers each side of joint)

But for efficient design the `AG´ for the gusset should be about equal to `AG´ of shell

Use properties of layup B without core

`AG´ shell/in. = 2.56 X $10^6$ X .14

= .358 X $10^6$ lb/in.

G for XP251-S, X-PLY = 2.4 X $10^6$ lb/in$^2$

$\therefore$ $t_{REQ} = \frac{.358}{2.4}$ = .15 i.e. 15 layers

For practical deisgn use 16 layers, i.e. .16" thick.

### 4.5.4 WING STIFFNESS

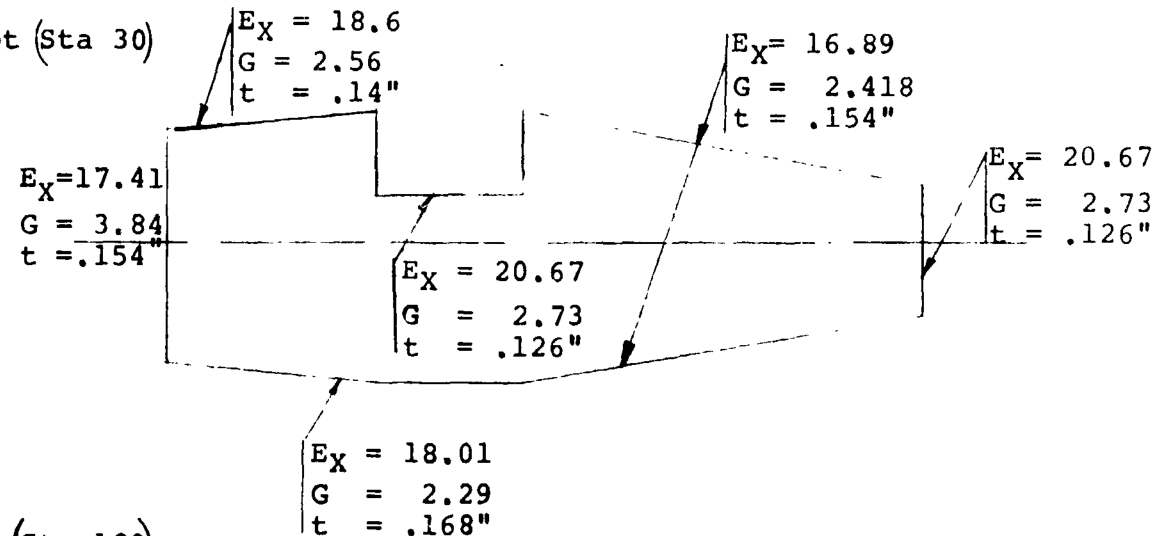
The effective values of EI and GJ are given by

$$(EI)_{EFF} = \sum_{1}^{n} E_i \ell_i t_i y_i^2 + \sum_{1}^{n} (E_i I_i)$$

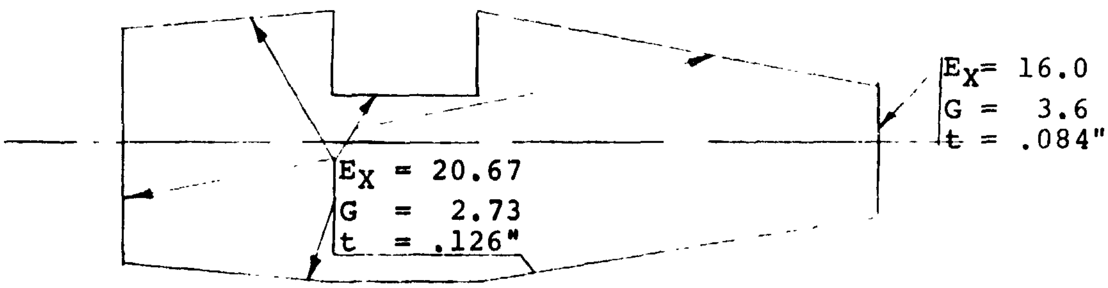
$$(GJ)_{EFF} = \frac{4A^2}{\sum_{1}^{n} \frac{\ell_i}{G_i t_i}}$$

The distribution of $E_X$, G and t for the root and tip sections are shown below.

Root (Sta 30)




Tip (Sta 180)




(Note: $E_X$ and G in $10^6$ lb/in$^2$ units)

The stiffness parameters at wing Station 30 and Station 180 are computed below.

| ITEM | $E_X \times 10^{-6}$ LB/IN.$^2$ | $\ell$ IN. | t IN. | y IN. | $E_X \ell t$ X $10^{-6}$ LB. | $E_X \ell t y$ X $10^{-6}$ LB.IN. | $y_i = y - \bar{y}$ IN. | $E_X \ell t y_i^2$ X $10^{-6}$ LB.IN.$^2$ | $E_X I_O$ X $10^{-6}$ LB.IN.$^2$ | G X $10^{-6}$ LB/IN.$^2$ | $\frac{\ell}{Gt}$ X $10^6$ IN.$^2$/LB. |
|---|---|---|---|---|---|---|---|---|---|---|---|
| 1 | 18.6 | 12.5 | .14 | 8 | 32.55 | 260.4 | 7.012 | 1600.4 | ----- | 2.56 | 34.877 |
| 2 | 20.67 | 4.87 | .126 | 5.93 | 12.68 | 75.21 | 4.942 | 309.7 | 25.21 | 2.73 | 14.157 |
| 3 | 20.67 | 7.5 | .126 | 3.5 | 19.53 | 68.37 | 2.512 | 123.1 | ----- | 2.73 | 21.80 |
| 4 | 20.67 | 4.75 | .126 | 5.88 | 12.37 | 72.74 | 4.892 | 296.0 | 23.24 | 2.73 | 13.81 |
| 5 | 16.89 | 22.25 | .154 | 6.5 | 57.87 | 376.18 | 5.512 | 1758.2 | ----- | 2.418 | 59.752 |
| 6 | 20.67 | 4.16 | .126 | 2.08 | 10.83 | 22.53 | 1.092 | 12.9 | 15.64 | 2.73 | 12.094 |
| 7 | 20.67 | 2.78 | .126 | -1.39 | 7.24 | -10.06 | -2.378 | 40.9 | 4.67 | 2.73 | 8.081 |
| 8 | 16.89 | 22.25 | .154 | -4.5 | 57.87 | -260.43 | -5.488 | 1742.9 | ----- | 2.418 | 59.752 |
| 9 | 18.01 | 7.5 | .168 | -5.62 | 22.69 | -127.53 | -6.608 | 990.8 | ----- | 2.29 | 19.495 |
| 10 | 18.01 | 12.5 | .168 | -5.58 | 37.82 | -211.04 | -6.568 | 1631.5 | ----- | 2.29 | 32.491 |
| 11 | 17.41 | 4.86 | .154 | -2.43 | 13.03 | -31.66 | -3.418 | 152.2 | 25.76 | 3.84 | 8.218 |
| 12 | 17.41 | 6.95 | .154 | 3.475 | 18.63 | 64.75 | 2.487 | 115.2 | 77.65 | 3.84 | 11.75 |
| | | | Σ | .988 | 303.11 | 299.46 | | 8733.8 | 172.17 | | 296.277 |
| 1 | 20.67 | 12.5 | .126 | 8 | 32.55 | 260.4 | 7.0 | 1595.0 | ----- | 2.73 | 36.339 |
| 2 | | 4.87 | | 5.93 | 12.61 | 74.75 | 4.93 | 306.5 | 25.14 | | 14.157 |
| 3 | | 7.5 | | 3.5 | 19.53 | 68.37 | 2.5 | 122.1 | ----- | | 21.803 |
| 4 | | 4.75 | | 5.88 | 12.37 | 72.73 | 4.88 | 294.6 | 23.24 | | 13.808 |
| 5 | 20.67 | 22.25 | .126 | 6.5 | 57.95 | 376.66 | 5.5 | 1753.0 | ----- | 2.73 | 64.684 |
| 6 | 16.0 | 4.16 | .084 | 2.08 | 5.59 | 11.63 | 1.08 | 6.5 | 8.06 | 3.6 | 13.756 |
| 7 | 16.0 | 2.78 | .084 | -1.39 | 3.74 | -5.93 | -2.39 | 21.4 | 2.41 | 3.6 | 9.193 |
| 8 | 20.67 | 22.25 | .126 | -4.5 | 57.95 | -260.78 | -5.5 | 1753.0 | ----- | 2.73 | 64.684 |
| 9 | | 7.5 | | -5.62 | 19.53 | -109.76 | -6.62 | 856.0 | ----- | | 21.803 |
| 10 | | 12.5 | | -5.58 | 32.55 | -181.63 | -6.58 | 1409.3 | ----- | | 36.339 |
| 11 | | 4.86 | | -2.43 | 12.66 | -30.76 | -3.43 | 148.9 | 25.03 | | 14.128 |
| 12 | 20.67 | 6.95 | .126 | 3.475 | 18.10 | 62.90 | 2.475 | 110.9 | 75.41 | 2.73 | 20.205 |
| | | | Σ | 1.000 | 285.13 | 285.21 | | 8377.2 | 159.29 | | 330.899 |

From the tabulated data, stiffness at root section (Station 30)

$$(EI)_X \text{ EFFECTIVE} = (8773.8 + 172.2) \times 10^6$$

$$= 8946.0 \times 10^6 \text{ LB IN.}^2$$

$$(EI) \text{ REQUIRED} = 7960 \times 10^6 \text{ LB IN.}^2$$

$$(GJ) \text{ EFFECTIVE} = \frac{4 \times 471^2}{296.277 \times 10^{-6}}$$

$$= 2995 \times 10^6 \text{ LB IN.}^2$$

$$(GJ) \text{ REQUIRED} = 1600 \text{ LB IN.}^2$$

Similarly at tip section (Station 180)

$$(EI)_X \text{ EFFECTIVE} = 8536.5 \times 10^6 \text{ LB IN.}^2$$

$$(EI) \text{ REQUIRED} = 6450 \times 10^6 \text{ LB IN.}^2$$

$$(GJ) \text{ EFFECTIVE} = \frac{4 \times 471^2}{330.899} = 2682 \times 10^6 \text{ LB IN.}^2$$

$$(GJ) \text{ REQUIRED} = 1600 \times 10^6 \text{ LB IN.}^2$$

Hence, design is satisfactory.

## 4.6 WEIGHTS SUMMARY - COMPOSITE WING FOR SAR AIRCRAFT

| ITEMS | WEIGHT (LB) |
|---|---|
| • COVERS | 415 |
| • CORE | 50 |
| • JOINTS & GUSSETS | 67 |
| • RIBS | 40 |
| • ADHESIVES | 24 |
| • NACELLE ATTACHMENT STRUCTURE | 80 |
| • LEADING & TRAILING EDGES, FAIRINGS | 169 |
| • SPLICES, FASTENERS, MISC. | 30 |
| TOTAL WING WEIGHT PER AIRCRAFT | 875 |

## 5.0 DEVELOPMENT PLAN

Presented in this section is a development plan for the incorporation of a composite wing box on an existing NASA tilt rotor research aircraft. Cost and schedules were developed on the basis of modifying an existing aircraft following completion of its flight test program.

### 5.1 DESIGN

The conceptual design of the main spar torque box would be identical to that described in the preceding section for the SAR aircraft. For the research aircraft, the wing could be resized to meet the existing NASA aircraft, and to minimize cost only the main torque box would be built of composites. The existing metal auxiliary surfaces would be attached to the composite spar. Fabricating auxiliary surfaces from composites has been demonstrated on previous projects so that demonstration of the spar box itself would be the prime objective of the program.

The design effort would mainly consist of the establishment of a final design for the composite wing box with provisions for interfacing systems and attached components (i.e., flaperons, spoilers, etc.) and fuselage attachment from an existing tilt rotor aircraft.

The following GFE components from the existing tilt rotor aircraft, assumed to be available in the 1978 time frame, would be installed in the wing:

- Engines
- Nacelles
- Shafting
- Surface controls - L.E. umbrellas, flaperons, spoilers
- Controls
- Transmissions
- Tilt mechanism

## 5.2 FABRICATION AND ASSEMBLY

The manufacturing effort includes the fabrication of test specimens, attachment fittings, tooling, a tool proving wing box and a flight wing box.

The Government supplied hardware from an assumed existing tilt rotor aircraft would be installed in the wing and final wing assembly would be accomplished at the contractor's facility.

Following the completion of the flight wing assembly with research instrumentation installed and calibrated, it will be shipped to NASA for installation on the existing tilt rotor research aircraft.

## 5.3 TESTS AND EVALUATION

### 5.3.1 Bench Tests

The bench test effort includes the design and fabrication of test fixtures, instrumentation and calibration of test specimens, performance of tests and preparation of test reports.

The type of tests planned are:

- Coupons - crack propagation and fatigue
- Panels - compression and shear
- Joints - tension and shear
- Adhesive compatibility
- Wing section - ultimate load
- Full scale wing - proof and dynamic shake
- Wing root attachment - proof
- Tool proving
- Environmental

### 5.3.2 Ground and Flight Tests

The ground and flight test program included in this estimate consists of the following:

- Proof load controls
- System functionals
- Dynamic shake
- Safety of flight review

. Pre-flight checks

. Helicopter mode

. Transition and fixed wing mode

The time span considered from shipment of wing assembly through flight tests is approximately 6 months. Boeing Vertol's effort during this period is in support of NASA personnel who will install the wing on the aircraft per Boeing furnished instructions and perform the necessary ground and flight checks.

The planning costs presented are based on projected CY 1977 planning dollars which is intended to represent an average for the period of performance.

Cost and schedule data comprising pages 5-5 to 5-8 has been removed from this volume since it is considered proprietary information to the Boeing Vertol Company.

## 6.0 REFERENCES

1. "V/STOL Tilt Rotor Aircraft Study, Volume I, Conceptual Design of Useful Military and/or Commercial Aircraft", NASA CR-114437, Boeing Company, Vertol Division, March 1972.

2. "V/STOL Tilt Rotor Aircraft Study, Volume II, Preliminary Design of Research Aircraft", NASA CR-114438, Boeing Company, Vertol Division, March 1972.

3. "V/STOL Tilt Rotor Aircraft Study, Volume III, Overall Research Aircraft Project Plan, Schedules, and Estimated Cost", NASA CR-114439, Boeing Company, Vertol Division, March 1972.

4. "V/STOL Tilt Rotor Aircraft Study, Volume IV Wind Tunnel Investigation Plan for a Full-Scale Tilt-Rotor Research Aircraft", NASA CR-114440, Boeing Company, Vertol Division, March 1972.

5. "Model 222 Structural Design Criteria", Boeing Document D222-10029-1.

6. Taylar, R. J.: "Weight Prediction Techniques and Trends for Composite Material Structures", SAWE Paper No. 887, presented at the 30th annual conference of the Society of Aeronautical Weight Engineers, Inc., May 1971.

7. Gabriel, E., "Drag Estimation of V/STOL Aircraft", Boeing Document D8-2194-1, Boeing Company, Vertol Division, May 1969.

8. Schoen, A. H., "User's Manual for VASCOMP II, The V/STOL Aircraft Sizing and Performance Computer Program", Boeing Document D8-0375, Volume VI, March 1968 (Revised October 1971).

9. Wisniewski, John S., "Weight Trends Data for VASCOMP", Boeing Document D8-0375-2, 1967.